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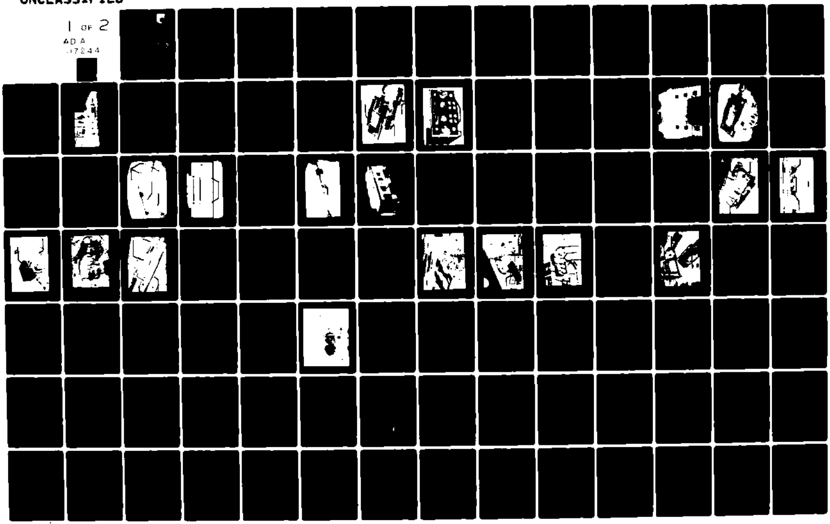

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AFWAL-TR-82-3035

FLIGHT TEST OF A G. E. AND DCI DIRECT DRIVE
FLY-BY-WIRE FLIGHT CONTROL SYSTEM

Gavin D. Jenney
Harry W. Schreadley
Carl N. Allbright

DYNAMIC CONTROLS, INC.
DAYTON, OHIO 45424

June 1982

Final Report for Period October 1978 - April 1981

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Gregory J. Cecere

GREGORY J. CECERE
Program Manager

Evard H Flinn

EVARD H. FLINN, Chief
Control Systems Development Branch
Flight Control Division

FOR THE COMMANDER

Ernest F. Moore

ERNEST F. MOORE, Col, USAF
Chief, Flight Control Division
Flight Dynamics Laboratory

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This report describes the flight test of two Fly-By-Wire flight control systems which use direct drive control valves. The test aircraft is an F-4E and the test systems are used to control the left aileron of the test aircraft. One test system was developed by the General Electric Company and the other by Dynamic Controls, Inc.
The report describes the test systems, the aircraft installation, flight test and results.

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FOREWORD

The effort described in this document was performed by Dynamic Controls, Inc. of Dayton, Ohio under Air Force Contract F33615-78-C-3609. The contract was performed under Project Number 1987, entitled "Flight Test Demonstration of Direct Drive Control Valve for Fly-By-Wire Flight Control Systems". Work under the contract was carried out in the Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio and at Edwards Air Force Base, California using Government facilities. The work was administered by Mr. Thomas Lewis and Mr. Gregory Cecere, AFWAL/FIGL Project Engineers.

This report covers work performed between October 1978 and April 1981. The technical report was submitted by the authors in March 1982.

The authors express their appreciation to Dynamic Controls, Inc. analyst, Heinrich J. Wieg for his contribution in the areas of design. The authors also express their appreciation to the program test pilot Captain Daniel Nims and flight test engineer Captain David Hatfield for their contributions to the program.

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LIST OF ABBREVIATIONS

A/C	aircraft
AFFTC	Air Force Flight Test Center
AOA	angle of attack
BLC	boundry layer control
CADC	Central Air Data Computer
CMD	command
DCI	Dynamic Controls, Incorporated
DDCV	Direct Drive Control Valve
EAFB	Edwards Air Force Base
EMI	electromagnetic interface
FBY	Fly-By-Wire
FCF	functional check flight
G. E.	General Electric
IFF	indentification system
KIAS	knots indicated airspeed
LED	light emitting diodes
LT	left
LVDT	linear variable differential transformer
P	Pilot (front seat)
RT	right
SAS	stability augmented systems
SW	switch
WPAFB	Wright-Patterson Air Force Base
WSO	Weapons Systems Officer (rear seat)

SECTION I

INTRODUCTION

1.1 General

This report describes the flight test of two different Direct Drive Control Valve Fly-By-Wire control system mechanizations in an F-4E aircraft. The Fly-By-Wire control systems were used to control an aileron of the test aircraft.

The purpose of the flight test program was to verify satisfactory performance of the Direct Drive Fly-By-Wire approach when used to control a fighter-bomber aircraft.

The flight test hardware evaluated consisted of two systems, one manufactured by the General Electric Company, Binghamton, New York and the other manufactured by Dynamic Controls, Inc., Dayton, Ohio. The systems were tested sequentially.

The flight test of the two systems was conducted at Edwards Air Force Base, California by the Air Force Flight Test Center (AFFTC). Interface hardware and flight test support were provided by Dynamic Controls, Inc.

This report describes in the following order:

(a) The Systems Evaluated
(b) The Test System Interface
(c) Aircraft Installations
(d) Flight Test of the General Electric System
(e) Flight Test of the Dynamic Controls, Inc. System
(f) Program Results

1.2 Background

The use of electronic primary flight control for aircraft, commonly called Fly-By-Wire (FBW), has brought with it the advantages of precise control and flexibility associated with electronic technology. However, the requirement for preservation of control with component failures has led to the use of parallel redundancy levels which are considerably greater than the conventional hydromechanical controls. This is in part due to lack of statistical confidence in the components used in a FBW system (since historical data on FBW flight control systems is not particularly abundant). Present FBW mechanizations for primary flight controls use up to four parallel channels of signal transmission in addition to three or four electrohydraulic channels in converting electronic commands to a hydromechanical output. The majority of the systems developed incorporate redundancy for the electrohydraulics by using a secondary actuator since the size and power requirements of the power actuator make incorporating three or four channel redundancy in the power actuator impractical.

Reducing the redundancy level of a FBW system to the two parallel channel single-fail-operate level of conventional controls, while meeting aircraft reliability requirements, may be practical with the state-of-the-art technology. If the number of components of the FBW system can be reduced and developed for maximum reliability, the reliability requirement for a single fail-operate FBW mechanization can be met.

The Direct Drive Control Valve (DDCV) accomplishes the simplification of the FBW system for a single fail-operate mechanization. For this type of valve, no hydraulic amplification stage is required. The electromechanical force generator is connected directly to the control spool used to control hydraulic flow to the power actuator. This type of mechanization eliminates the secondary actuator from the FBW mechanization.

This valve concept has been investigated by several companies, specifically by North American Rockwell (Columbus Division), General Electric Company (Johnson City), and Dynamic Controls, Incorporated. The type of force generator selected for development by each company has been different. North American Rockwell investigated a torque motor configuration. General Electric developed a linear force motor. Dynamic Controls, Inc. developed a moving coil force motor.

The flight test described in this report is the evaluation of the hardware resulting from Air Force funded direct drive development programs by General Electric and Dynamic Controls, Inc.

1.3 Program Summary

Installation and flight test of the two Direct Drive Control Valve (DDCV) Fly-By-Wire (FBW) systems occurred during the period from 8 October 1980 to 14 April 1981 at Edwards Air Force Base, California. The systems were installed in F-4E Aircraft No. 287. The General Electric Company (G. E.) system was installed and evaluated first. Figure 1 shows the test aircraft during installation of the DDCV systems.

Both DDCV FBW systems were used to control the left aileron of the test aircraft. The test hardware was designed to match the original aileron's performance characteristics. This was done in order to maintain symetrical control characteristics in roll, since only one aileron was changed to FBW control.

The DDCV mechanizations evaluated were similar in concept, but different in execution. The G. E. mechanization used a linear force motor attached to the normal aileron actuator's control valve and linear variable differential transformers (LVDT's) for position measurement. The Dynamic Controls, Inc. (DCI) mechanization used voice coil force motors attached to each end of a control valve which replaced the normal aileron control valve and housing. Linear potentiometers (direct current) were used in the DCI mechanization for position measurement. The G. E. mechanization required both 28 volt DC and 115 volt 400 Hz AC electrical power for operation. The DCI system required 28 volt DC power for its operation.

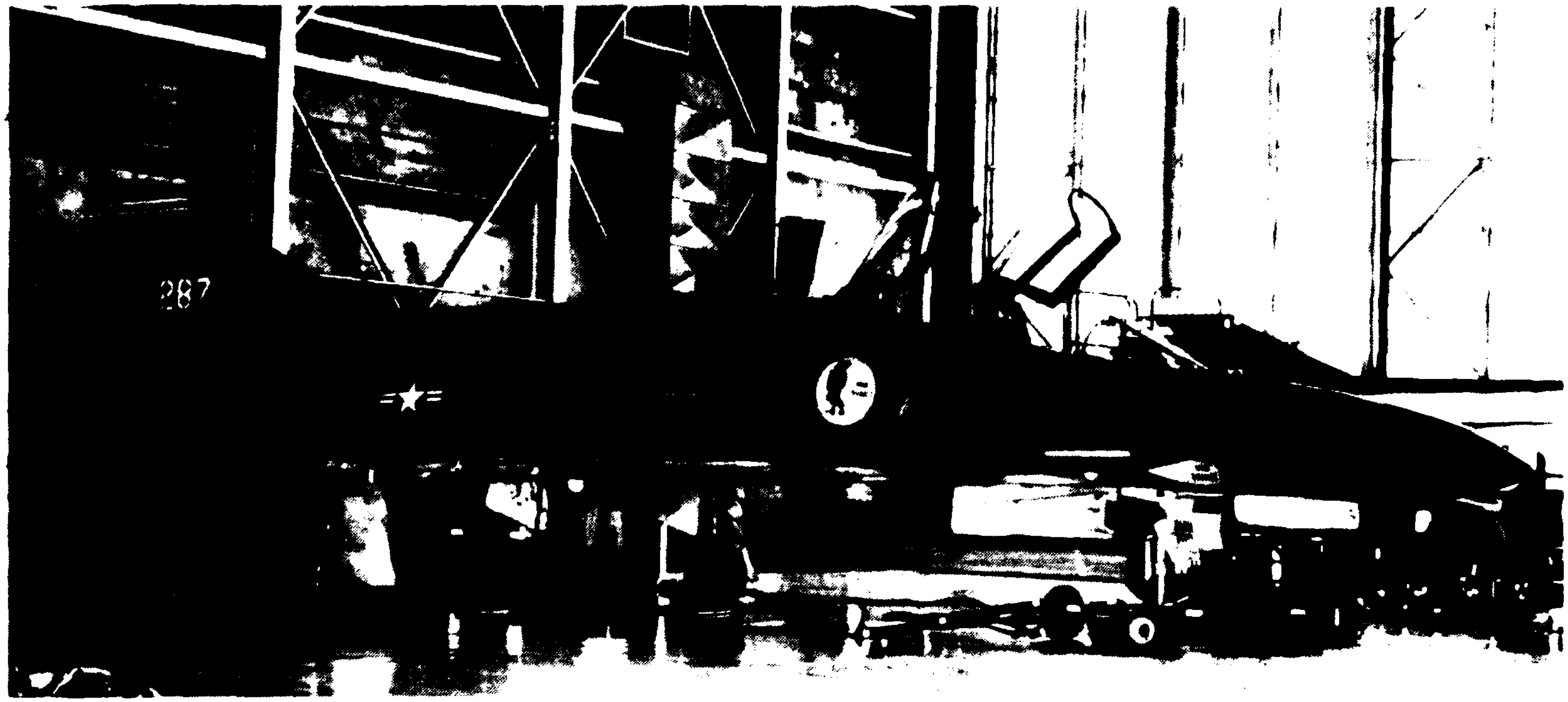


Figure 1   F-4E Test Aircraft

Interface hardware used in mounting the DDCV electronic assemblies for both systems was fabricated by DCI and flightworthiness tested prior to start of the installation into the aircraft. A common wiring harness for control was used for both systems. Adapter cables were used to interface each individual system to the common harness.

As originally scheduled with Edwards Air Force Base, the installation and flight test period was to be from June 1979 to June 1980. The aircraft initially assigned was F-4E aircraft No. 368. However, prior to start of installation, aircraft 368 became unavailable. The test aircraft was then changed to No. 287 and because of the assignment of higher priority programs to the aircraft the length of time for installation and test reduced from twelve months to six months. This change impacted directly upon the flight time obtained for each system. (The installation time originally scheduled was the same as the total installation and flight test time for the new aircraft assignment.) The period from 8 October 1980 to 20 February 1981 was used for installation and flight test of the G. E. system. The remaining seven weeks to 14 April 1981 were used to install and flight test the DCI system. Due to both incidential maintenance requirements on the test aircraft and the length of time required to install and wring-out the test systems, the flight test time obtained on each test system was less than the 25 hours per system originally planned. However, both systems were successfully flight tested over most of the planned test envelope and exhibited performance characteristics similar to the normal aircraft aileron control system. Although the limited flight test time did not indicate long term reliability characteristics, the program did demonstrate satisfactory flight performance of the DDCV mechanization for both systems.

SECTION II

DIRECT DRIVE SYSTEMS DESCRIPTION

2.1 General

The General Electric Company and the Dynamic Controls, Incorporated Direct Drive Control Valve FBW mechanizations were designed and developed independently, under separate Air Force contracts. Both systems are basically the same in concept, using fail-operate (two channel) redundancy with a single stage servovalve driven by electrical force motors. The force motors of both mechanizations have four coils (two per channel) and four corresponding current drivers. Both mechanizations were designed to be applied to the left aileron control of the F-4 aircraft, changing the normal actuator from a mechanical input actuator to FBW control. Both mechanizations are based on modification of the normal F-4 actuator control valve assembly and the addition of electrical feedback position transducers to the normal actuator.

The General Electric Company mechanization is described in detail in the technical report, AFFDL-TR-78-32. The Dynamic Controls, Inc. mechanization is described in detail in the technical report AFFDL-TR-77-91. The following material in this section is a summary description of the two DDCV mechanizations.

2.2 General Electric System Description

Figure 2 is a block diagram of the General Electric system. The system is designed using a force motor that has a moving armature and four driving coils which are driven by individual servoamplifiers (two coils and two servoamplifiers per channel). The force motor has two inner loop position feedback LVDT's (one per channel) with two independent outputs which are set up out of phase in order to implement a self-monitoring scheme.

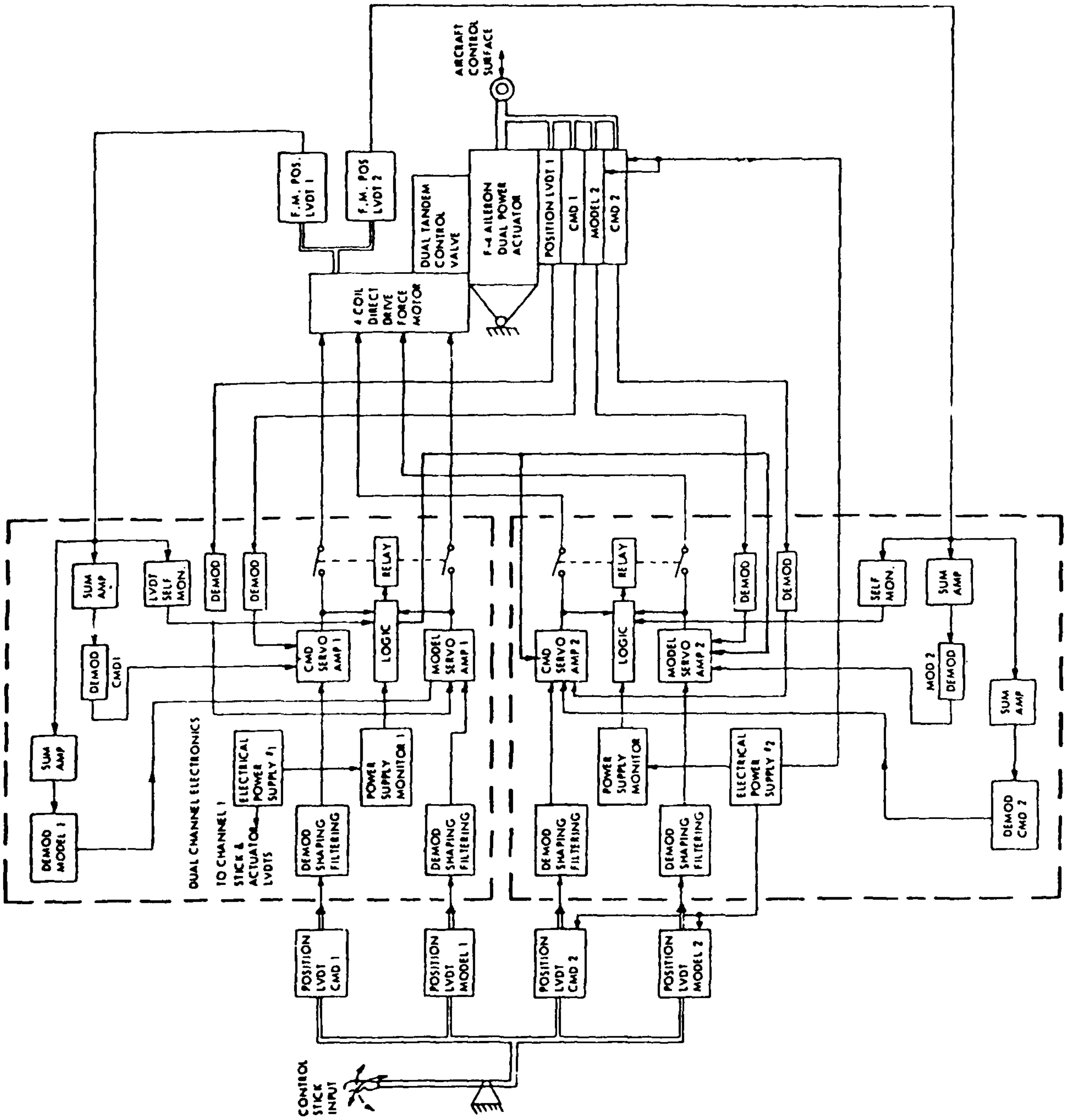


Figure 2   G. E. DDCV FBW System

There are two command and two actuator position LVDT's per channel. Within each channel one set of command and feedback transducers are summed at the CMD servoamplifier and the other set of transducers are summed at the model servoamplifier. The error signal developed at each servoamplifier drives the independent coils of the force motor within the respective control channels.

The two channels have a cross strap from Channel 1 failure logic to Channel 2 CMD and model servoamplifiers. The cross strap acts as a gain changer allowing Channel 1 to act as a master channel until Channel 1 fails, (or both channels have large error signals) then Channel 2 switches to normal system gain. Should Channel 2 fail, there is no effect on Channel 1 output.

The failure logic monitors the status of each channel's power supply, servoamplifier error voltage, force motor current and force motor position LVDT output. The failure logic disconnects the servoamplifiers from the force motor coils and provides a failure indication when the signals being monitored exceed their disagreement limits.

Figure 3 shows the G. E. Direct Drive Flight Test Actuator. The force motor is attached to the manual input control valve normally used on an F-4 aileron actuator. The position transducers measuring the actuator output position are mounted in the center depression of the actuator body.

Figure 4 shows the G. E. control electronics used with the direct drive actuator. A single box contains both the Channel 1 and the Channel 2 electronics.

2.3 Dynamic Controls, Inc. System Description

Figure 5 is a block diagram of the Dynamic Controls, Inc. control system. The control system is a fail-operate configuration (as is the G. E. system) where any single hydraulic or electrical failure does not prevent the control system from operating with a satisfactory level of performance. The

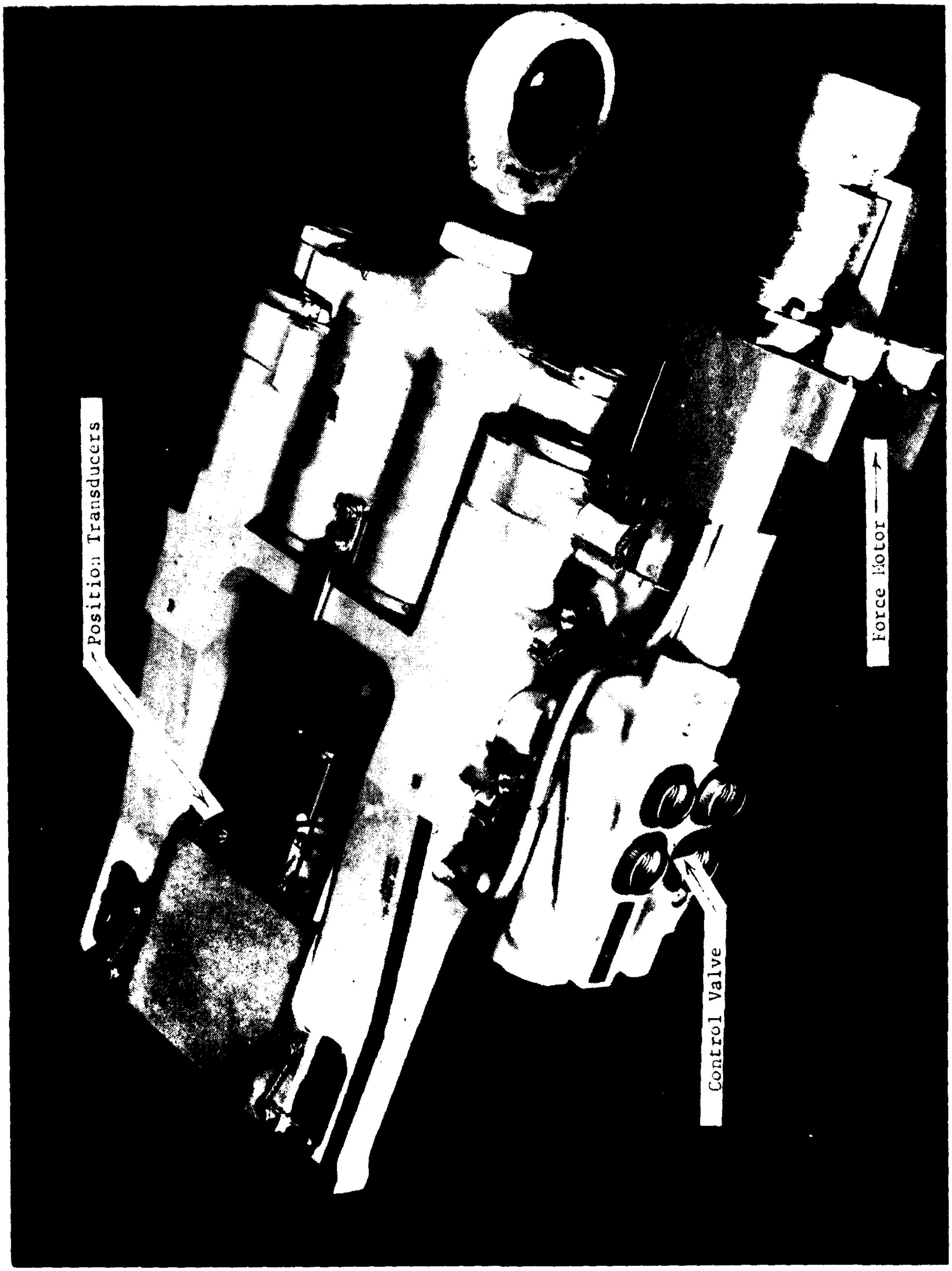


Figure 3 G. E. Direct Drive Flight Test Actuator

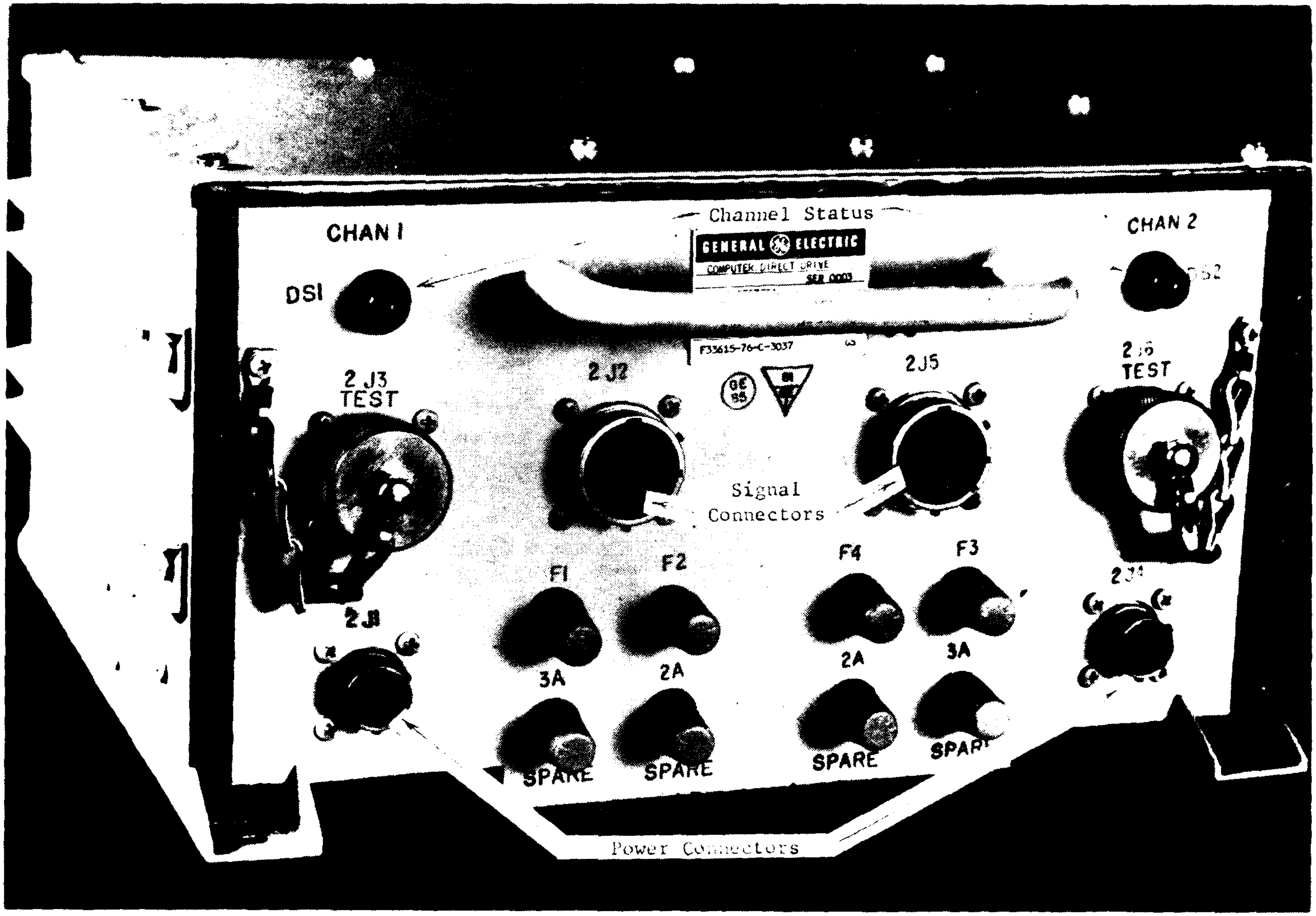


Figure 4 G. E. Control Electronics

system uses two self-monitored control channels and a two section electro-hydraulic control actuator. The control actuator uses two force motors connected directly to the tandem control spool which meters hydraulic flow to the actuator drive areas.

In normal operating mode, both self-monitoring control channels operate, driving both moving coil force motors. The power spool is spring centered. Either control channel can apply sufficient driving force to the tandem spool to achieve maximum stoke. The two self-monitored control channels are completely independent (both mechanically and electrically) down to the tandem power spool. Channel independence is necessary to eliminate the possibility of a common mode failure of the two control channels.

For the system to meet the "fail-operate" criteria, hardover failure of a control channel must be prevented. Hardover failures of command and feedback elements are detected and removed from affecting the channel output by using two monitor command and feedback elements for each control channel. As shown in Figure 5, command input 2 and feedback 2 are used as the monitor for command input 1 and feedback 1. The monitor feedback and command element outputs are compared with the output of the command and feedback elements used to control the servoamplifiers connected to the force motor. The comparison is made after the summing junction for the command and feedback signals. Disagreement between the command and monitor portions of each control channel causes the command portion to be disconnected from control of the servoamplifiers. To prevent hardover channel outputs due to servoamplifier failure, two servoamplifiers and two force motor coils are used in each channel. The servoamplifiers are cross-strapped in their current feedback paths so that failure of one amplifier-coil section are offset by the output of the other section. Four independent electrical power supplies and two hydraulic power supplies must be used to allow the system shown in Figure 5 to accept single electrical or hydraulic failures and continue to operate.

Figure 6 shows the electrical components of the DCI system. The electronic circuits are constructed as modules in order to provide fail-operate capa-

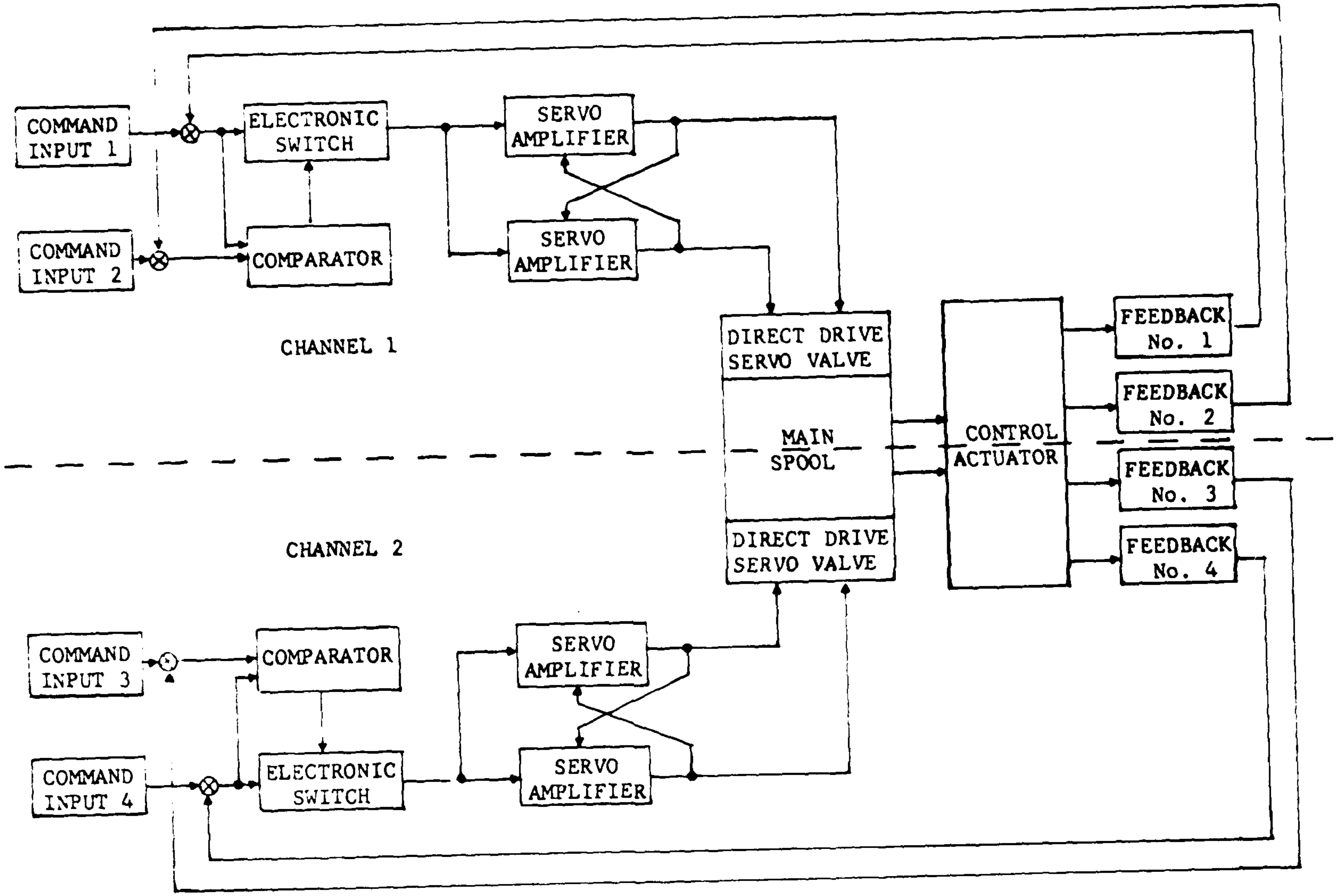
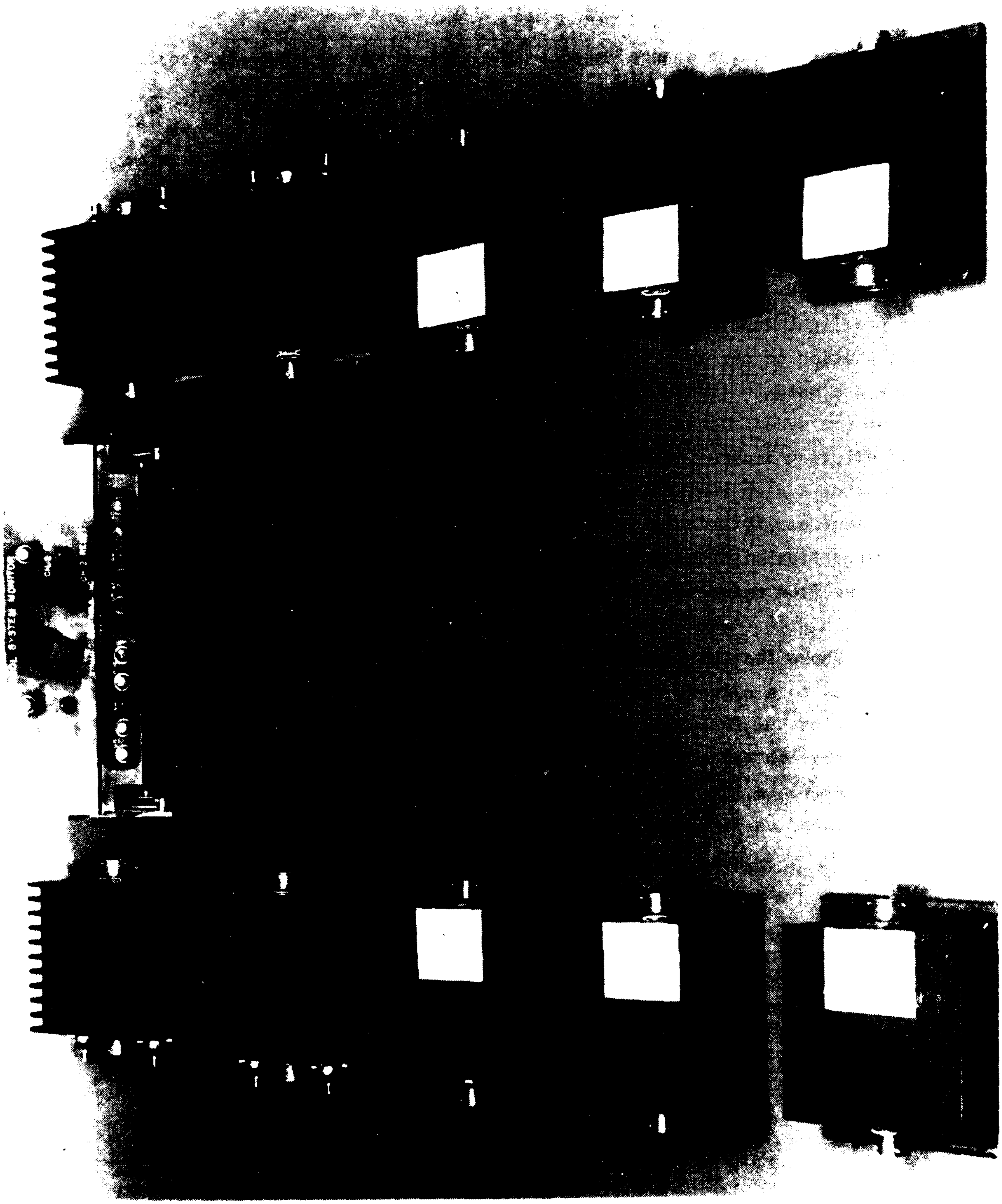


Figure 5 Dynamic Controls, Inc. Direct Drive Control System

bility after a single weapon impact. The size of the control electronics is primarily determined by this single hit survivability requirement used for the hardware design. The system electronics use a total of 16 operational amplifiers for control and failure detection. The modules with cooling fins are the servoamplifiers used to control the force motors. Included as part of the system is a control system monitor module. The control system monitor module uses an additional 8 operational amplifiers as light drivers for the display lights. Besides in-flight status display, the module allows preflight functional checking of each servoamplifier's operation and the operation of the failure monitor and disconnect circuits. Fiber optic cables are used to transmit status information from the servoamplifiers to the control system monitor module. The fiber optics are connected between light emitting diodes (LED'S) in the servoamplifiers and light drivers in the control system monitor module. The LED drivers are designed to illuminate when a servoamplifier's output reaches a particular level (for each direction of current flow). Fiber optics are used to prevent failure of the status display lines from affecting the functioning of the control system.

Figure 7 shows the DCI Direct Drive Flight Test Actuator. The force motors are mounted to a control valve body which replaces the normal manual input control valve. A force motor is used at each end of the control valve. Channel 1 of the control electronics is connected to one force motor and Channel 2 is connected to the other. The position potentiometers are mounted in the center depression of the actuator body.



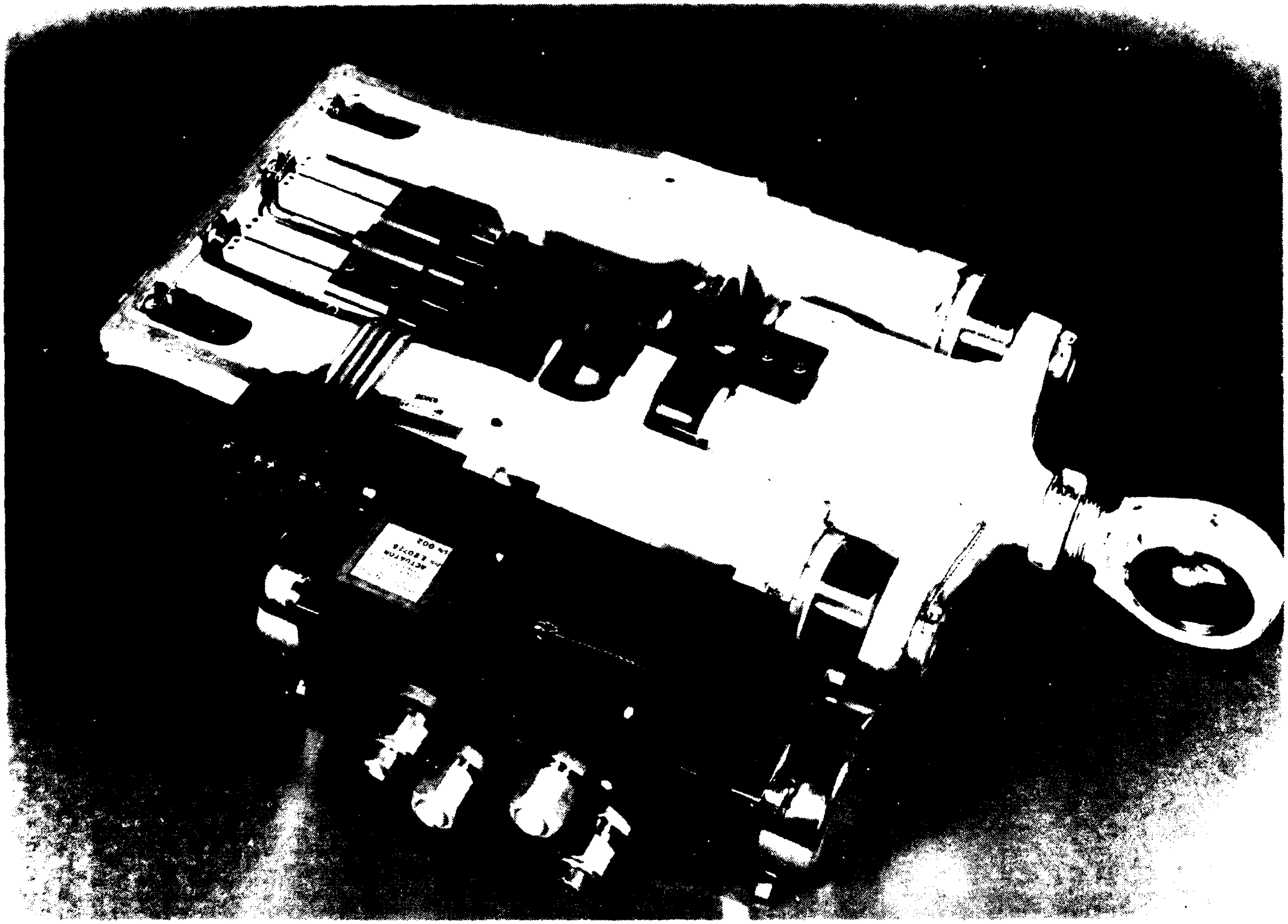


Figure 7 DCI Direct Drive Flight Test Actuator

SECTION III

DESCRIPTION OF THE TEST SYSTEM AIRCRAFT INTERFACE

3.1 General Approach

The purpose of installing the DDCV FBW systems in the F-4 aircraft was to demonstrate the feasibility of utilizing that type of control system in an aircraft's primary flight control system. The selection of the aileron axis for the location of the test systems was made early in the hardware development programs for both systems. The philosophy of using a single aileron control channel is that a hardover failure of a test system could be compensated with the normal aircraft spoilers and the opposite aileron. The "E" model of the F-4 aircraft was recommended by Edwards AFB as the test aircraft because that particular model of the F-4 has controllable slotted leading edges on the wings, allowing a larger angle of attack before loss of control.

The direct drive test hardware was designed to match the normal aileron hardware performance. Therefore, the criteria for success of the DDCV FBW systems was that they provide performance with no significant difference from that of a normal F-4 aircraft aileron system.

Normally, installation of a FBW system as a replacement for a hydromechanical control system will provide improved control sensitivity by eliminating hysteresis and deadband induced by the mechanical linkages. However, for the test system (since FBW was applied to only one of the ailerons of the test aircraft) the command transducers for the test systems were mounted at the end of the normal mechanical linkage chain. This mounting choice was made to retain the same sensitivity characteristics in both the right and left roll modes and to allow the surface trim and autopilot mechanization to function normally.

The two test systems, although functionally similar, were mechanized with sufficient differences that direct interchangeability of individual components was not possible. In designing the Class II modification for the test aircraft, a common wiring harness was used. Short adapter cables were used (where required) to couple specific components of a system to the harness. This wiring approach minimized the effort required to change from one test system to the other. Both systems included a control system monitor module, control electronics, a DDCV actuator and a command transducer. Figure 8 shows the location of these components in the test aircraft. The DDCV actuator for both systems replaced the normal aileron actuator in the left wing. The command transducer was mounted adjacent to the DDCV actuator, at the end of the control linkage chain. This location for the command transducer provided the normal trim and automatic flight control system inputs to the left aileron actuator. The control system electronics for both systems was installed in the upper equipment bay behind the rear seat. Electrical power connection to the appropriate aircraft supply voltages (115 volts @ 400 Hz and 28 VDC for the G. E. system, 28 VDC for the DCI system) was made through relays and circuit breakers mounted in the rear seat cockpit. The control system monitor module was installed in the right-hand console of the rear seat cockpit.

3.2 G. E. System Interface Hardware

Two components of the test system shown in Figure 8 were fabricated and flightworthiness tested by DCI as part of the flight test interface for the G. E. system. These components were the command transducer shown in Figure 9 and the control system monitor shown in Figure 10. The G. E. command transducer housed four LVDT's manufactured by Pickering, Inc. to the same specification used by G. E. for the actuator position transducers. Excitation and demodulation of the command transducers' output was provided in the G. E. electronics package. Figure 9 also shows the mounting bracket and a link used to install the command transducer in the test aircraft. (Only one of the two brackets shown in Figure 9 was used in the aircraft installation.)

The pilot monitor was constructed to allow preflight test of the G. E. system by the rear seat pilot using the system status indication and channel reset. As shown in Figure 10, the control system monitor used

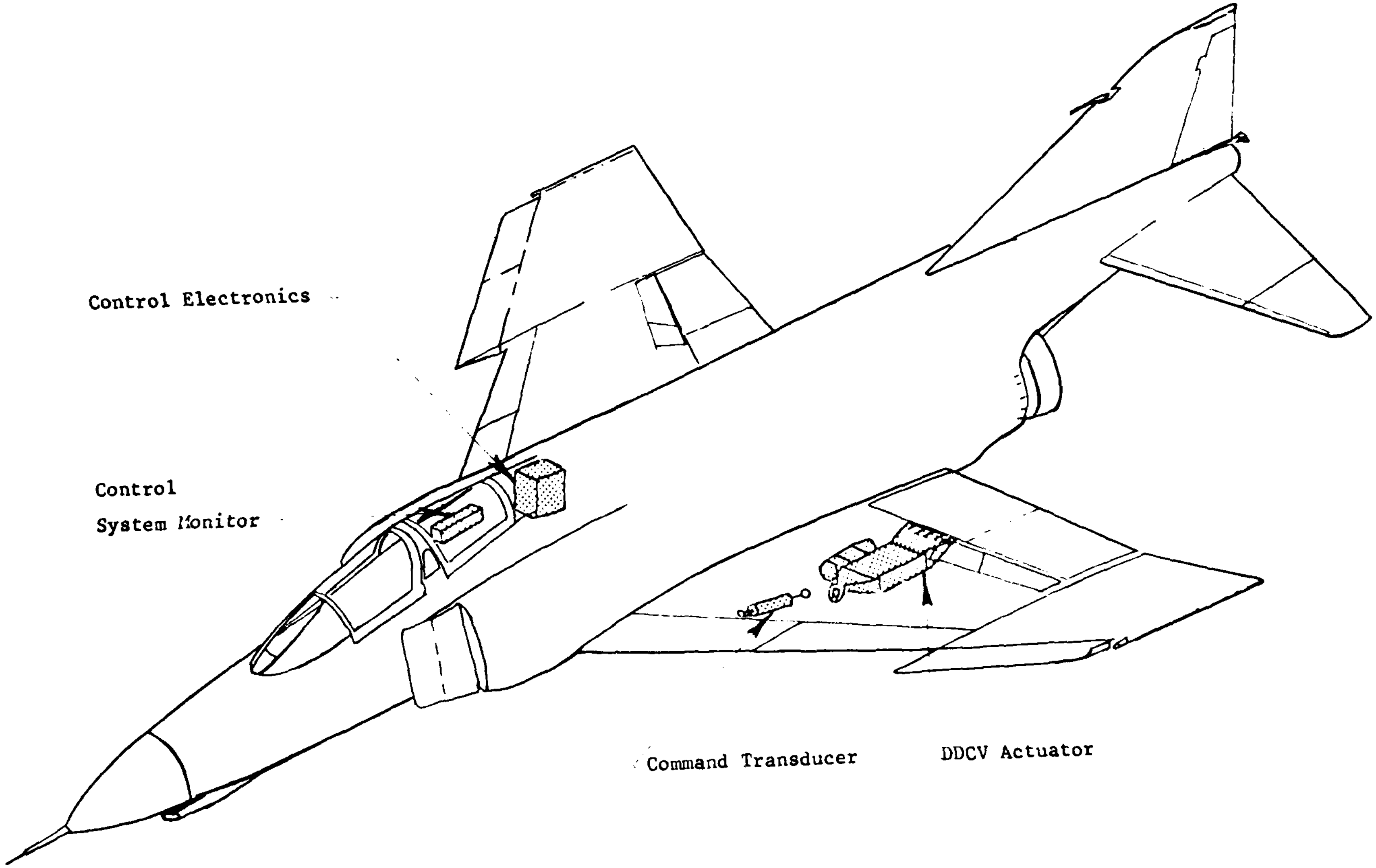


Figure 8 General Test System Installation

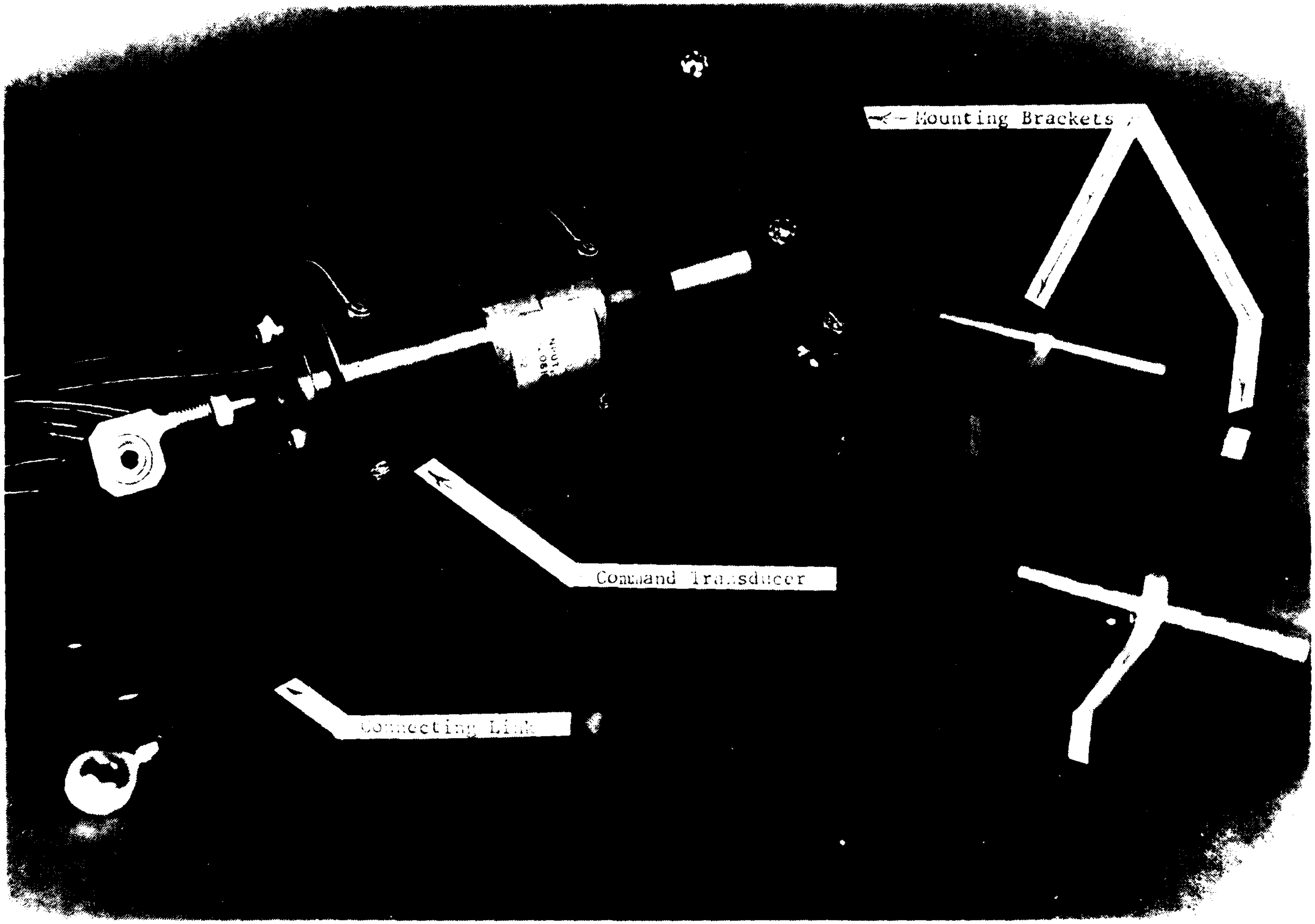


Figure 9 G. E. System Command Transducer

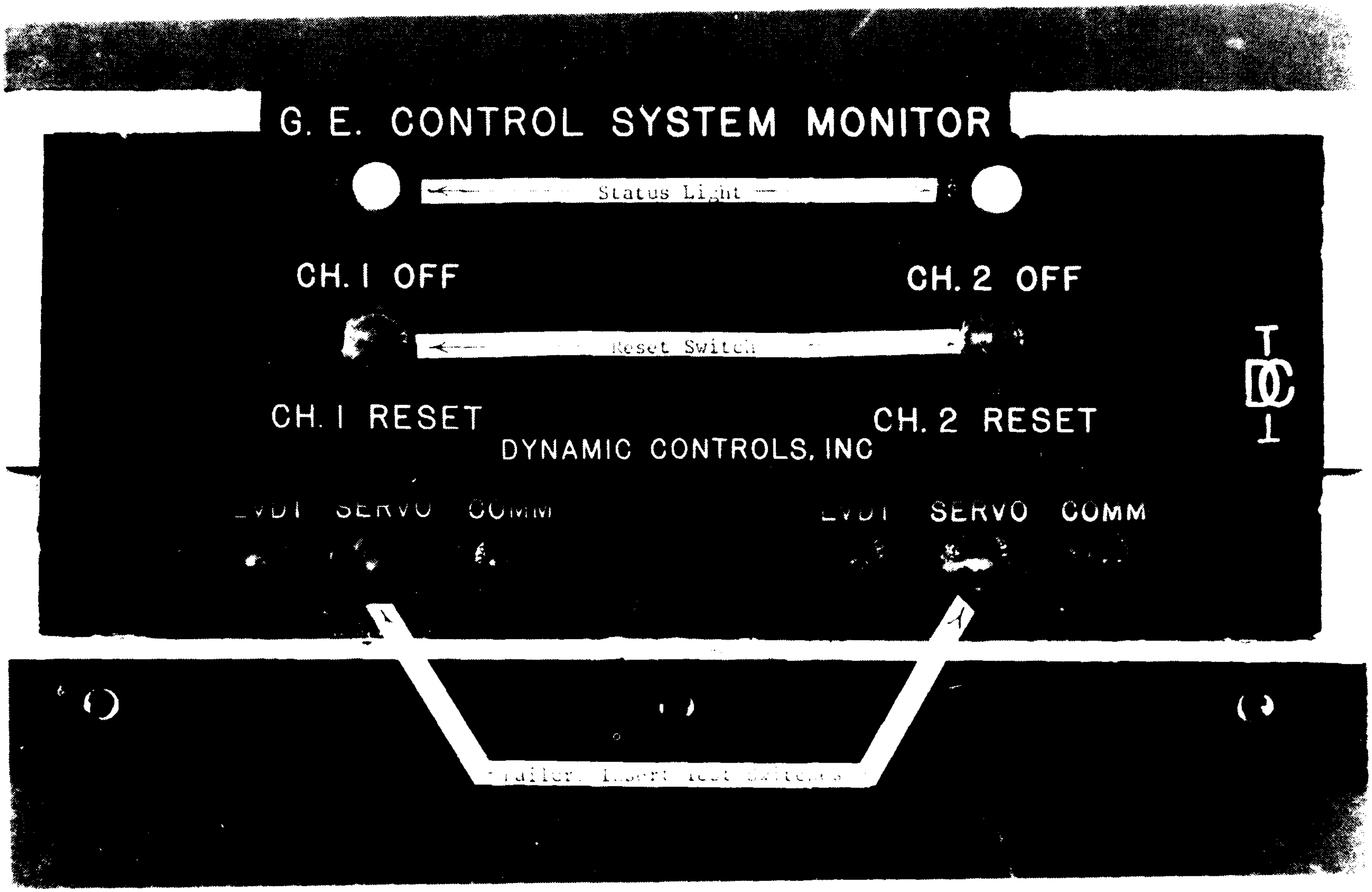


Figure 10 G. E. System Control System Monitor

three momentary contact switches for each channel to simulate electrical failure signals in the G. E. control electronics. These failure signals were used to preflight check the failure detection sections of the control electronics. The control system monitor also included a channel reset switch and a status light for each channel. Failure of a control channel illuminated that particular channel's "off" light.

3.3 DCI System Interface Hardware

The DCI system required two additional hardware components in order to interface with the test aircraft. These components were the command transducer and the power supply package. Figure 11 shows the command transducer. The transducer housed four cermet potentiometers, identical to those used for position feedback of the DCI DDCV actuator. Figure 11 shows the transducer without the silicon rubber accordian boot used over a portion of the outside housing for contamination protection. The transducer used vent holes in one face of the housing in order to allow air to enter the housing without inflating or deflating the protective boot. Air passing through these vent holes was filtered with fine mesh screen discs.

Figure 12 shows the power supply package which housed four DC to DC converters. Each converter used 28 volts DC as the power source and provided $\pm$ 15 volts DC as an output. Four converters were used in order to provide the necessary power supply independence for fail operate redundancy with a power supply failure. Figure 12 shows the power supply package with the side cover removed. The four servo-amplifiers of the DCI system were mounted on the top surface of the power supply for packaging convenience.

3.4 Interface Hardware Flightworthiness Testing

The basic flight control system hardware for the DDCV FBW systems (developed under previous Air Force contracts by G. E. and DCI) had been flightworthiness tested during development. However, as part of the flight test program it was necessary to flightworthiness test the following interface hardware manufactured by DCI for the program:

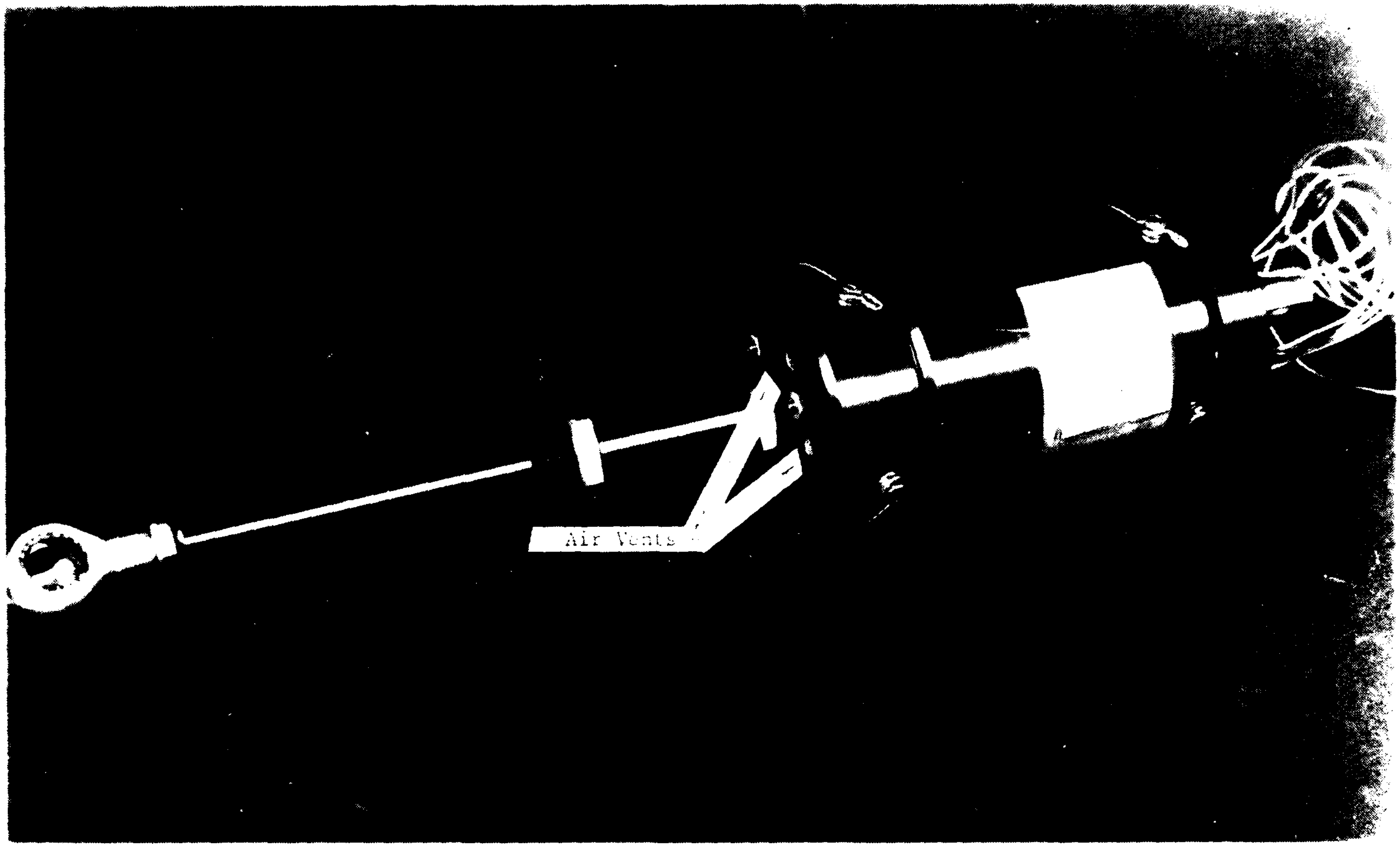


Figure 11 DCI System Command Transducer

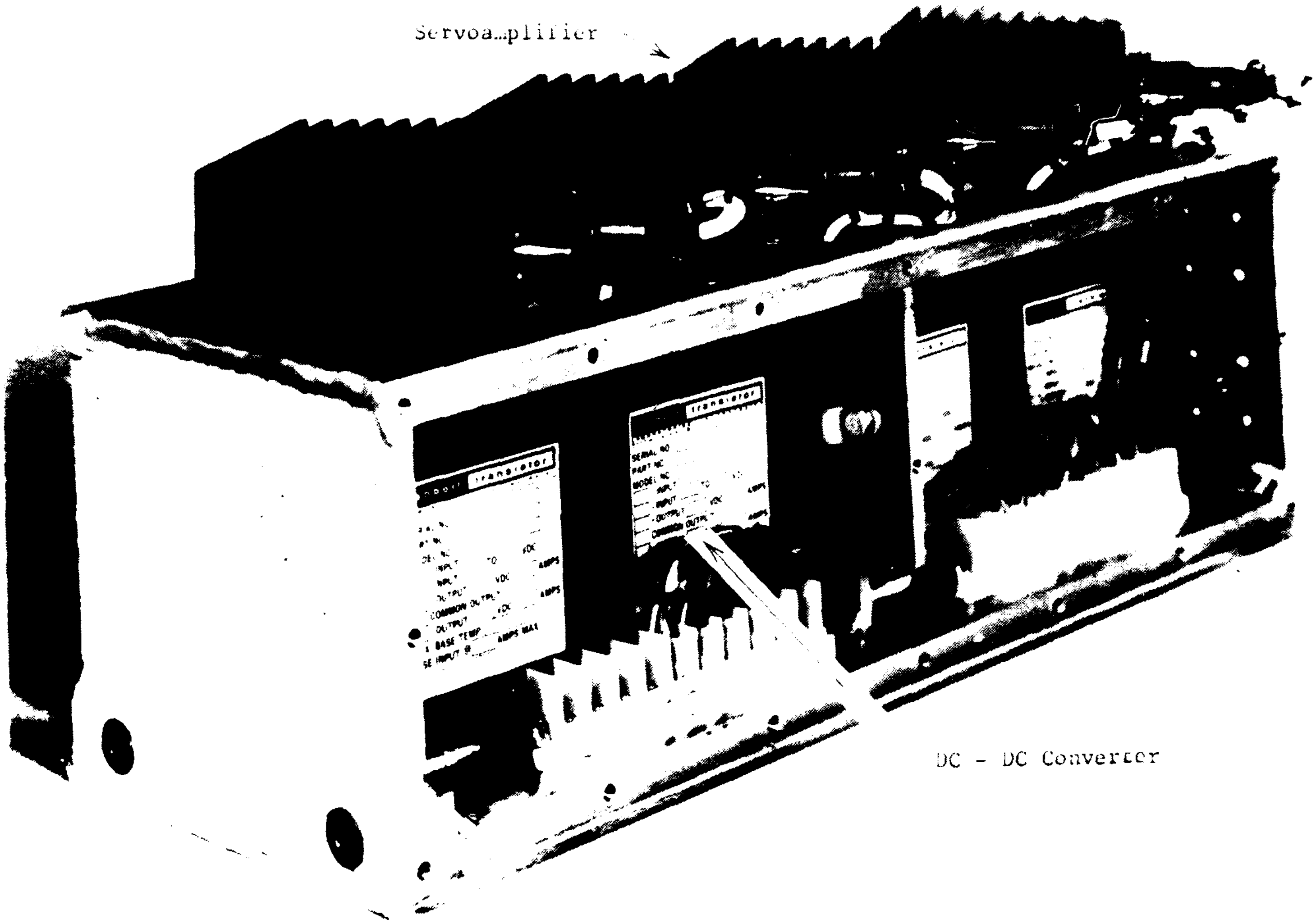


Figure 12 DCI System Interface Power Supply

1. The command transducer for the DCI DDCV system
2. The command transducer for the G. E. DDCV system
3. The control system monitor for the G. E. DDCV system
4. The power supply module for the DCI DDCV system

Items 1, 2, 3, and 4 above, were subjected to an Air Force approved sequence of low temperature, high temperature, shock and vibration tests taken from MIL-STD-810C and consistent with the testing used to certify the hardware of both DDCV systems. Two specimens of hardware for these items were fabricated, one set for flight use and the other for flightworthiness certification. For item 4 (since the converters used were purchased as flight qualified hardware) only the enclosure (with dummy masses substituted for the converters) was tested for vibration and snock.

With one exception, the interface hardware passed the initial flightworthiness testing sequence without difficulty. The exception was that the command transducer for the DCI system failed the input force level limit of 10 lbs at the low temperature of $-40^{o}F$. The failure to pass the low temperature force requirement was due to the boot used to protect the transducer from contaminants. Although rated for $-40^{o}F$ service, the neoprene material of the boot became too stiff at that temperature. The boot material was changed to silicon rubber and the transducer passed the low temperature test without difficulty.

3.5 Mass Change to the Test Aircraft with the Test System

The mass balance of the test aircraft was not changed significantly by the installation of either test system. The following is a list of the weights of the components of the two DDCV systems and the weights of the components removed from the installation locations in the aircraft.

TABLE 1 COMPONENT WEIGHTS

G. E. System

Unit Installed	Unit Removed
1. DDCV Aileron Actuator @ 58 lb	Standard Aileron Actuator @ 50 lb
2. Command Transducer and Brackets @ 5 lb	Standard Input Linkage @ 3 lb
3. Control Electronics @ 15 lb	ASQ-91 Weapons Release Computer @ 36 lb
4. Control System Monitor @ 3 lb	2 Blank Panels and Inertial Navigation Panel @ 3 lb

DCI System

Unit Installed	Unit Removed
1. DDCV Aileron Actuator @ 62 lb	Standard Aileron Actuator @ 50 lb
2. Command Transducer and Brackets @ 5 lb	Standard Input Linkage @ 3 lb
3. Power Supply Module and Servoamplifiers @ 32 lb plus Control Electronics (six modules) @ 6 lb	ASQ-91 Weapons Release Computer @ 36 lb
4. Control System Monitor @ 3 lb	2 Blank Panels and Inertial Navigation Panel @ 3 lb

In addition, there was approximately 40 lbs of wire and connectors installed in the aircraft for interconnecting cabling for both systems. Approximately 20 lbs of the wire cabling was distributed over the inboard 10 ft of the left wing.

3.6 The Power Interface - Hydraulic and Electrical

Both DDCV systems used the same hydraulic connections and had the same pressure/flow requirements as the standard aileron actuator. The electrical power required for the two systems was the following:

G. E. System	Quiescent		Maximum Slew
Channel 1	28 VDC	3 watts	3 watts
	115 VAC (400 Hz)	20 watts	180 watts
Channel 2	28 VDC	3 watts	3 watts
	115 VAC (400 Hz)	20 watts	180 watts
DCI System	**Quiescent**		**Maximum Slew**
Power Source 1	28 VDC	.8 watts	25 watts
Power Source 2	28 VDC	.8 watts	25 watts
Power Source 3	28 VDC	.8 watts	25 watts
Power Source 4	28 VDC	.8 watts	25 watts

The above power requirements include the excitation power required for the command transducers and control systems monitors. Both systems were connected to the aircraft electrical power through circuit breakers rated at 5 amperes. The G. E. System Channel 1 was connected to the 115 VAC left main buss and the 28 VDC main buss of the test aircraft. The G. E. System Channel 2 was connected to the 115 VAC right main buss and the 28 VDC essential buss of the test aircraft. The DCI Channels 1A and 1B were connected to the 28 VDC main buss of the test aircraft. The DCI Channels 2A and 2B were connected to the 28 VDC essential buss of the test aircraft.

SECTION IV

TEST SYSTEM INSTALLATION

4.1 General

In order to provide access to the installation area for test systems wiring, the left engine was removed. Test wiring was also routed from the left wing aileron compartment through the left wing, left engine nacelle and into the upper equipment bay. Test wiring was also routed from the upper equipment bay through the pressure bulkhead and into the rear right cockpit console. Test system power was routed from circuit breakers above the rear right hand cockpit console, to the rear cockpit instrument panel, and into the upper equipment bay through the pressure bulkhead. This wiring was used by both the G. E. and DCI systems.

In addition to the wiring required to operate the test systems, an electrical shutdown circuit (as recommended by the Edwards AFB Safety Review Board) was incorporated into the test system installation. This circuit provided the pilot and copilot a means of disconnecting the test systems from electrical power, thereby removing the test systems from control. Turning off the electrical power allowed the DDCV actuators to retract to a $+1^{\circ}$ up aileron surface position at a rate of approximately 3° per second. Both DDCV test actuator control valves were mechanically adjusted to provide the desired retract direction of motion and rate of the actuator at valve null. Initial installation of the electrical power disconnect was accomplished by using a 4 pole normally open relay. The control voltage for the relay coil was controlled by "paddle switches" already used in the normal aircraft for AFSC momentary disconnect and located just below the grip on the front and back seat control sticks.

To mount the electronics into the aircraft equipment bay (including instrumentation components) brackets designed by Edwards AFB were fabricated at Wright-Patterson AFB. Upon completion of the common wiring, the G. E. DDCV test system was installed as the first test system.

4.2 G. E. System Aircraft Installation

Figure 13 shows the G. E. DDCV actuator and input transducer as installed in the test aircraft. Note that the command transducer is mounted just forward of the actuator attachment clevis. Figure 14 shows the G. E. DDCV actuator installation as viewed looking forward toward the leading edge of the wing and with the spoiler surfaces raised to full deflection. Figures 15 and 16 show the aircraft installation of the G. E. flight computer and instrumentation electronics. Figure 15 shows the mounting rack removed from the upper equipment bay and Figure 17 shows the G. E. control system monitor installation.

4.3 G. E. System Ground Check-Out

Much of the G. E. system installation time was spent in ground operation and check-out of the system. This section lists significant problems encountered during the ground check-out and the solutions. This section also documents changes to the G. E. computer (described in detail in report AFFDL-TR-78-32) made during this time period. The problems and solutions are listed in chronological order as they occurred.

When electrical power was initially applied to the test systems with a 120 volt, 400 Hz ground powered generator, the 3 amp fuses in the G. E. computer opened (reference the front panel as shown in Figure 4). This was attributed to the additional load imposed by adding the excitation of four additional LVDT's in the command transducer and the voltage level of 120 volts (slightly higher than the nominal 110 volt design point for the system). The solution to this problem was to install fuses rated at 5 amperes in place of the original ones. No further opening of the fuses was experienced during the test program.

Figure 13 G. E. Actuator/Input Transducer

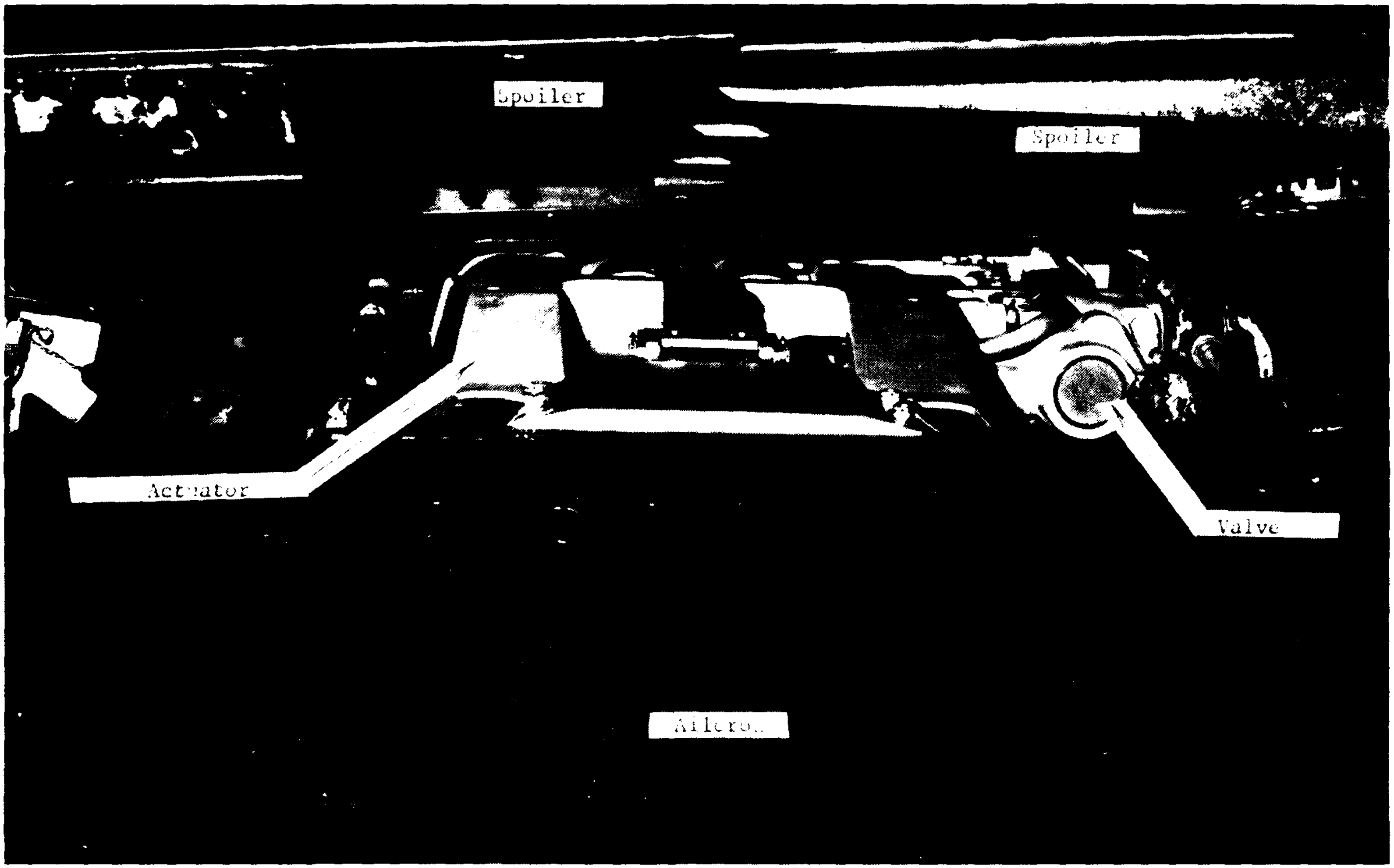


Figure 14 G. E. Actuator Installation

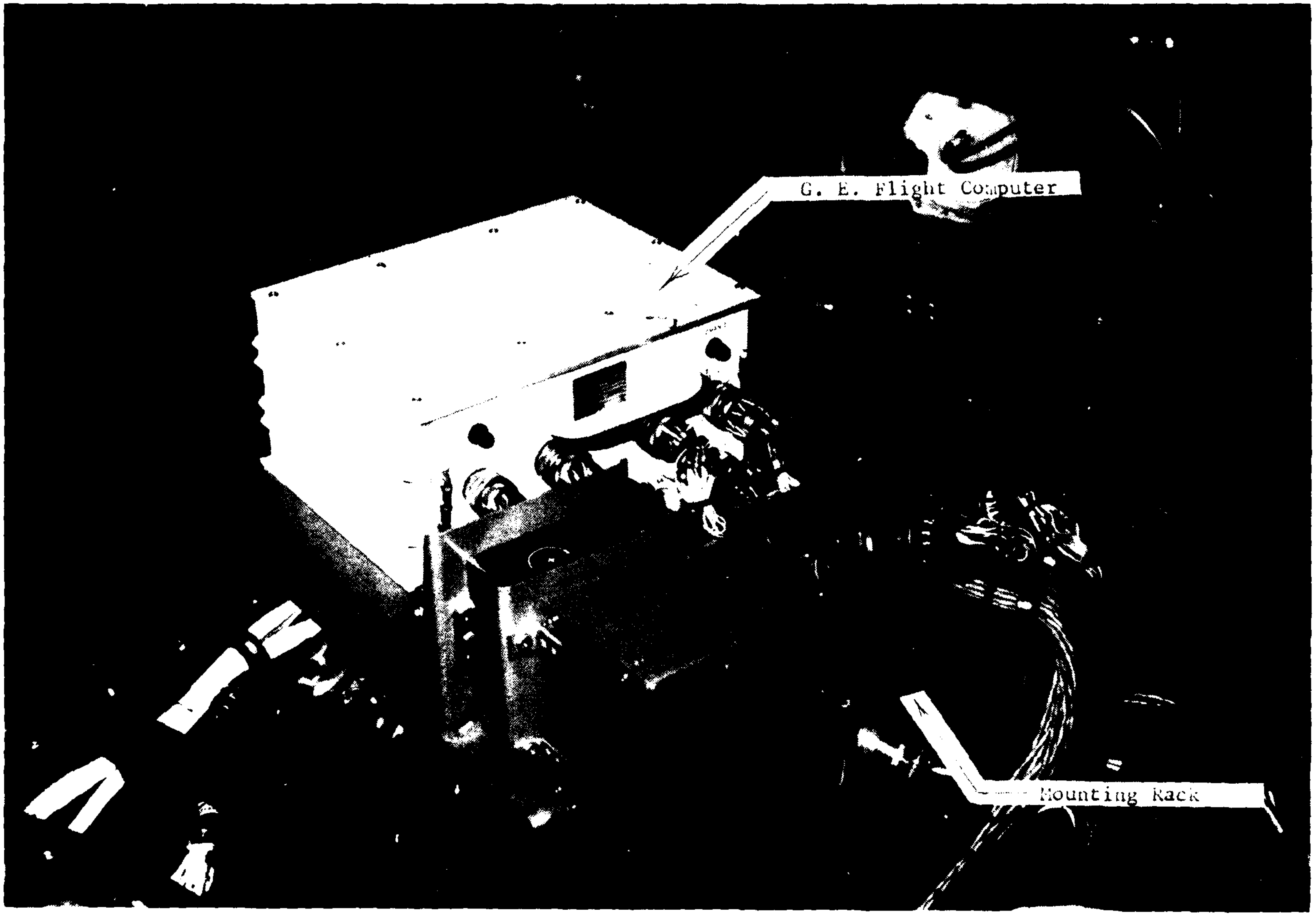


Figure 15   G. E. Electronics Mounting Configuration

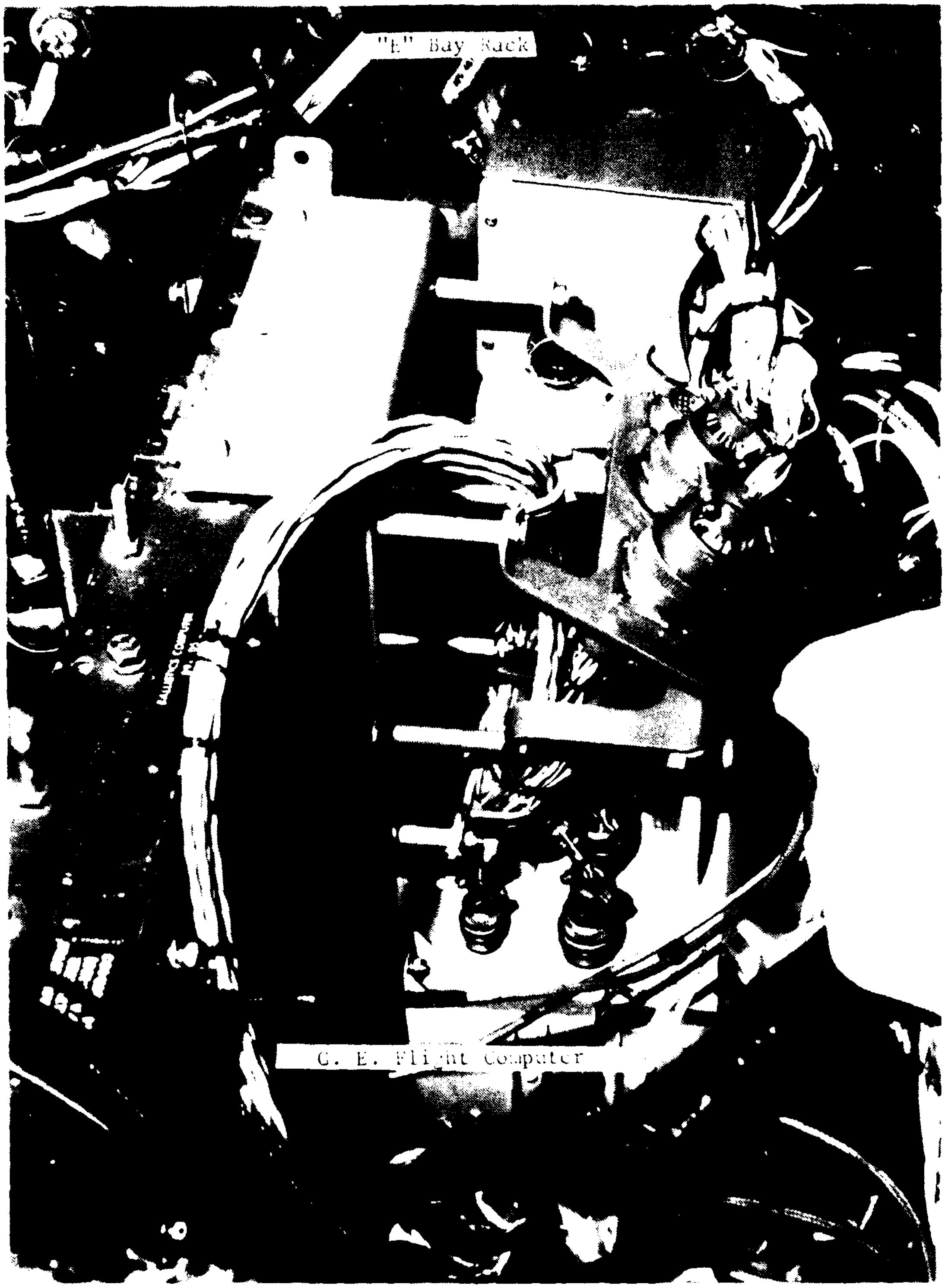


Figure 16 G. E. Flight Computer Installation

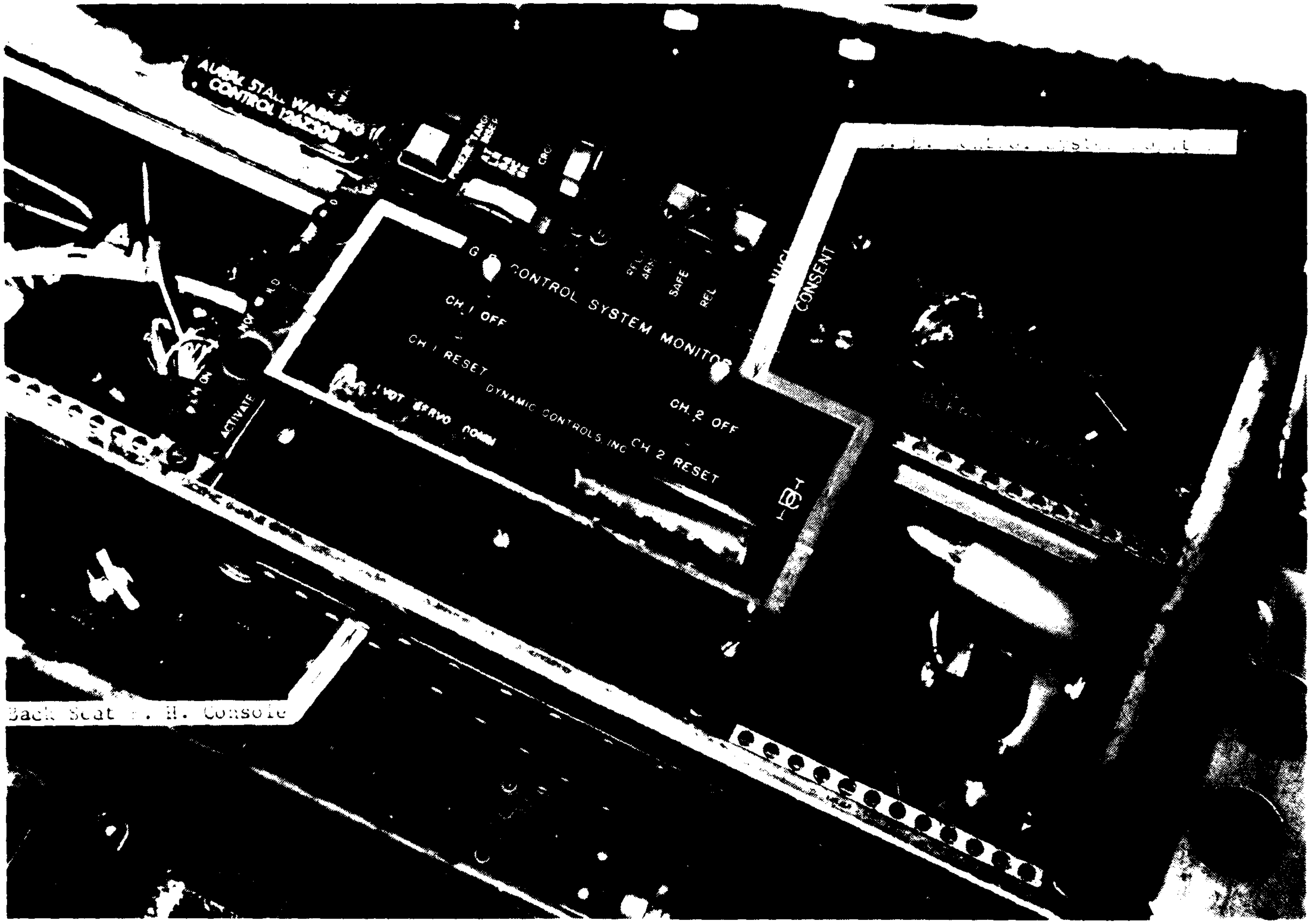


Figure 17 G. E. Control System Monitor Installation

When hydraulic power was initially applied to the system, the left aileron would not deflect more than 24 degrees with full stick input (the aircraft requirement is a deflection of 30 to 33 degrees). Therefore, the summing resistors R38 for the command input were changed from 4.75×10^3 ohms to 3.92×10^3 ohms in order to increase the input command signal authority. Figure 18 is the electrical schematic of one of the four control sections (two sections making up one channel) and the failure logic for one channel. The change to resistors R38 (and the other changes made during the ground check-out) are shown with a crosshatch superimposed.

A surface position transient of an amplitude going from 2 degrees up to 10 degrees down at maximum surface rate was experienced upon reset of Channels 1 or 2 after failure or of both channels upon start-up. Since inflight reset was one of the test program operational requirements, this transient was unacceptable. The cause of the transient was traced to a low pass filter constructed of capacitors C_4, C_5, and R79, placed in the feedback path of amplifier AR2 on circuit boards A1, A2, A4, and A5 (reference Figure 18). The reason for the transient was that, upon both channels failing, the actuator retracted and created a large offset voltage into the channel electronics (which are disconnected from control of the force motors until reset). This offset voltage caused the output of amplifiers AR2 to go to saturated output voltage, charging capacitors C_4 and C_5. Upon reset, this voltage discharged into the summing junction of amplifier AR2, causing the force motors to command the large transient motion of the aileron surface when either channel (or both) was reset. The solution to this problem was to install relays on circuit boards A1, A2, A4, and A5 with normally closed contacts shorting out C_4, C_5, and R37. This turned amplifier AR2 into a zero gain amplifier until the relays operated, preventing the charge build-up on the filter capacitors. The operating coils of the added relays were connected to the voltage source that operated existing relays K2 and K4 (which connected the driving electronics to the force motors).

A nuisance disconnect problem was encountered with the failure logic on circuit boards A3 and A6 of the G. E. flight computer (reference Figure 18) when the system was operated in the aircraft. The failure logic was

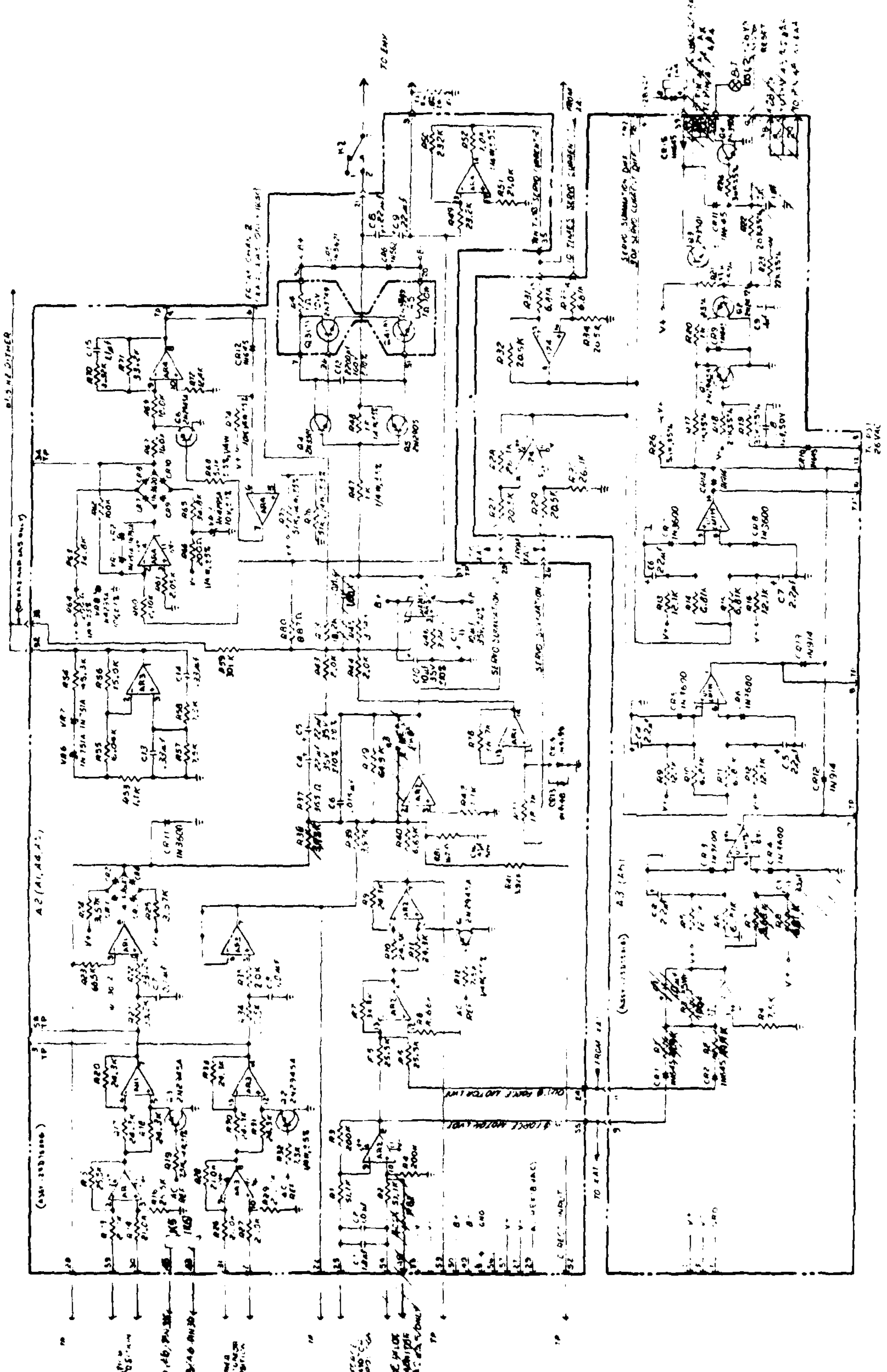


Figure 18 G. E. Control Electronics Schematic

designed to detect discrepancies of the force motor LVDT's, force motor servo error voltages, the current output of the amplifier stages connected to the force motor coils and power input voltages. Preflight check-out of the failure logic was accomplished by injecting failures into a channel by using the three failure inject switches located on the control system monitor (reference Figure 17). The problem encountered was a "nuisance disconnect" of one or both channels when exercising the system on the ground. The problem was traced to a thermal drift of the force motor LVDT outputs, apparently due to the heating caused by the 120 volt 400 Hz power used with the aircraft installation and reduced airflow. This voltage was higher than the nominal voltage of 110 volts used for the design of the electronics and used for bench check-out of the system at WPAFB, prior to the shipment of the hardware to Edwards AFB. The failure logic detection level for the force motor LVDT outputs was changed by DCI during the bench check-out because the bench operation at WPAFB showed that the logic as delivered to the Air Force was not sensitive enough to detect an open force motor LVDT coil. The failure detection level was set at that time to be equal to 50% of the maximum LVDT output voltage. The modification was made by reducing the gain of amplifier U2 on boards A3 and A6 (by changing resistors R1 and R2 from 21×10^3 ohms to 49.9×10^3 ohms). This allowed the control system monitor test switches to fail a channel when the force motor LVDT output was shorted to ground through a 20 ohm resistor. In the aircraft the 50% detection level was too sensitive to accept the LVDT output differences encountered with the 120 volt supply to the system. The solution to this problem was to increase the detection level to 60%. The change made by decreasing comparitor U1's input resistor R7 from 6.81×10^3 ohms to 3.65×10^3 ohms, input resistor R8 from 32.4×10^3 ohms to 6.81×10^3 ohms, deleting resistor R25 and connecting one end of resistor R6 to ground (reference Figure 18). At the same time the detection level was changed, the detection test circuit was modified by removing the 20 ohm shorting resistors from the LVDT outputs, attaching a 4.02×10^3 ohm resistor to the non-inverting input of amplifier AR-2 on boards A1 and A4. The "LVDT Fail" switch in the control system monitor grounded this resistor in the "test" mode.

During an engine run-up check on the flight line, Channel 1 failed and would not reset. Transistors Q1 and Q2 on the computer mainframe had failed at the base-to-collector junction and bias resistor R1 had failed mechanically (reference Figure 18). These components were replaced with parts from the spare computer. Upon check-out of the computer after the repair, it was discovered that the outputs of the four amplifiers driving the force motor coils for Channels 1 and 2 oscillated at 40 Hz at an amplitude of 30 volts peak to peak. Since the oscillation (perhaps caused by the aircraft wiring capacitance of the wire connecting the amplifiers to the actuator degrading the stability of the current amplifiers) could cause overheating of the output stages of the control channels, stabilizing capacitors were added to the computer. The oscillations were stabilized by adding .01 mfd capacitors across feedback resistors R45 on boards A1, A2, A4 and A5. This charged amplifier AR5 to a lag filter with a break frequency of 42.5 Hz.

During system check-out, it was noted that the LVDT fail switch required holding 4 seconds to fail Channels 1 or 2. This time was reduced by replacing the original 2.2 mfd capacitor C1 on boards A3 and A6 with a 1.0 mfd capacitor.

During ground check-out, it was discovered that with Channel 1 failed, operating the Channel 2 LVDT fail switch caused Channel 2 to fail and Channel 1 to reset at the same time. This characteristic was eliminated by adding a .1 mfd capacitor between the junctions of R22, R24, CR11 and ground on boards A3 and A6 providing transient rejection.

4.4 DCI System Aircraft Installation

Figure 19 shows the DCI DDCV actuator installed in the test aircraft. Figure 20 shows the DCI input transducer installation. Note the protective boot used with the transducer and that the transducer is mounted immediately forward of the DDCV actuator in the left wing. Figure 21 shows the electronics installation in the equipment bay behind the rear seat. Note the fiber optic cables used to connect the servoamplifier status indicating lights to the

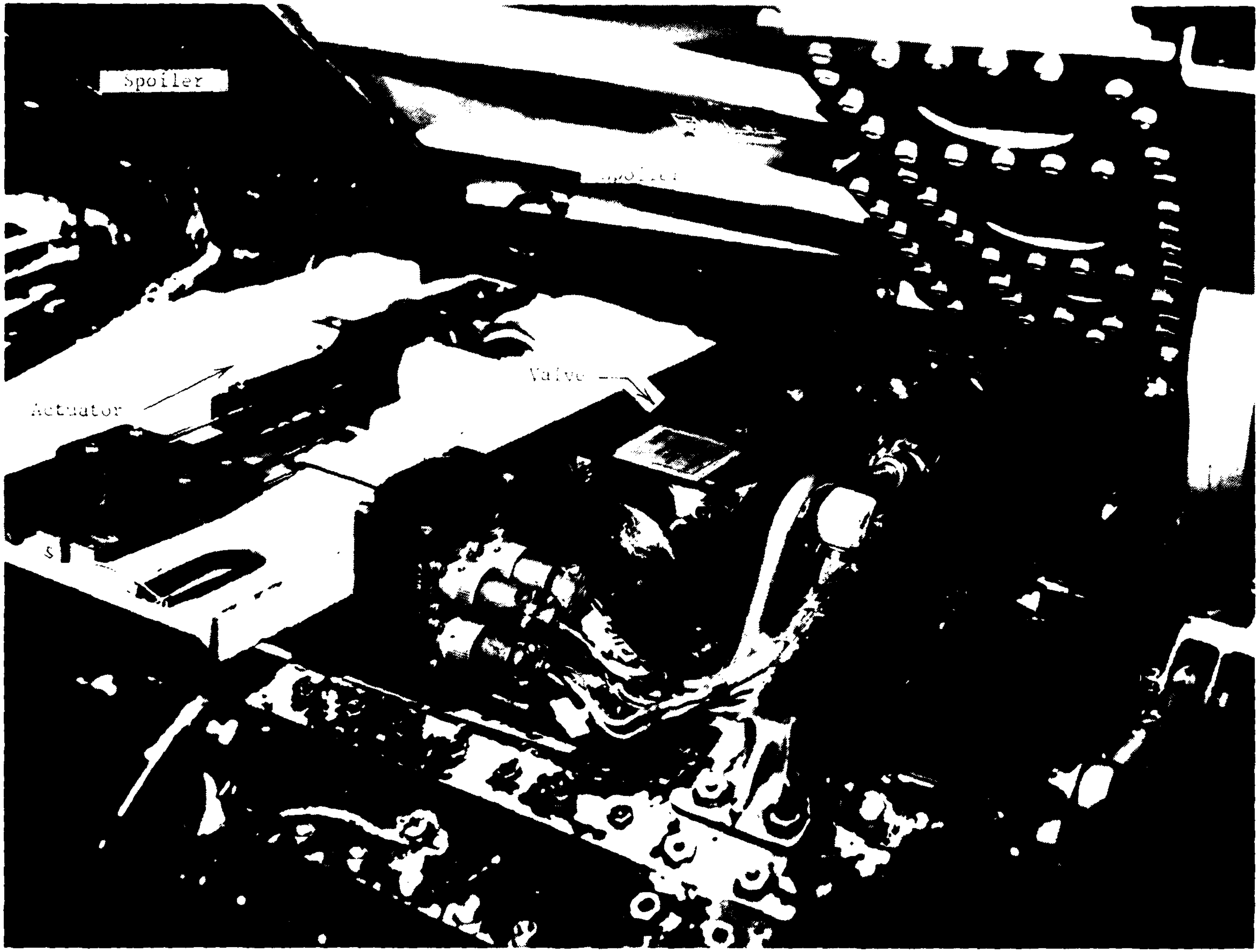


Figure 19 DCI Actuator Installation

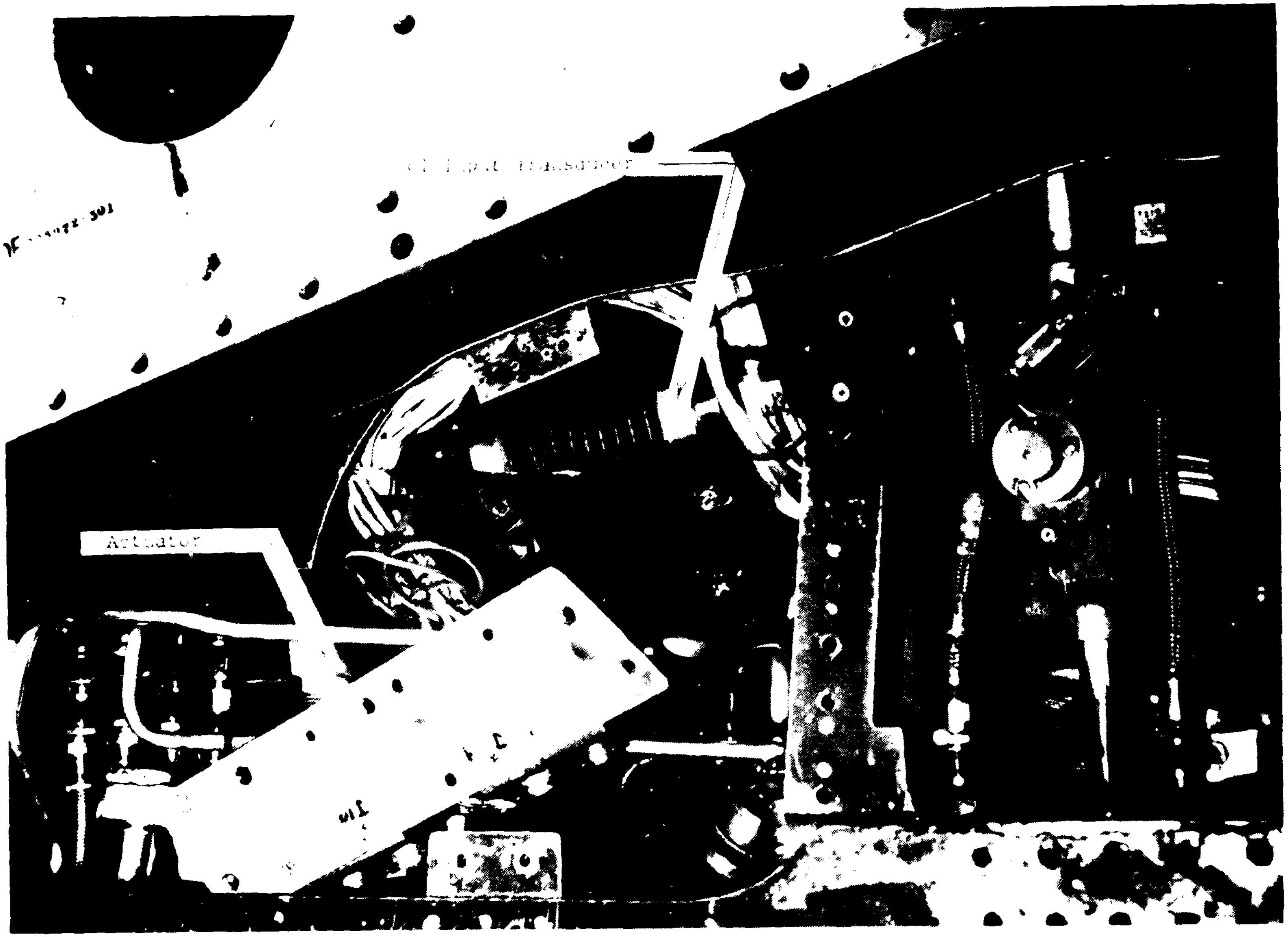


Figure 2[illegible] DCI Input Transducer Installation

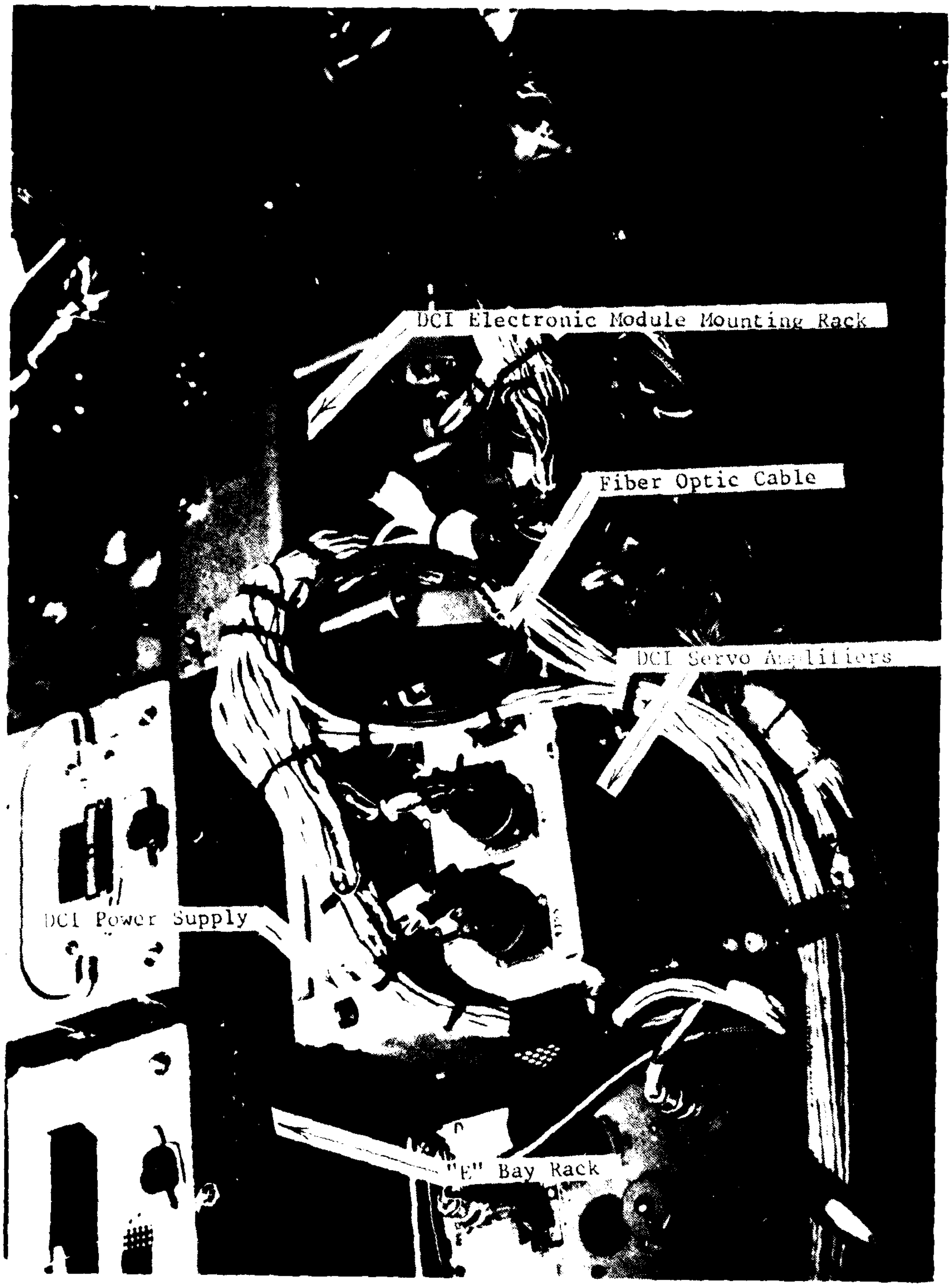


Figure 21 DCI Electronics Installation

control system monitor installed in the back seat cockpit. Figure 22 shows the DCI control system monitor installation in the right hand console of the back seat. Also shown in Figure 22 is the test system circuit breaker panel.

4.5 DCI Systems Ground Check-Out

Approximately six weeks were spent in installation and ground check-out of the DCI DDCV system. This section lists significant problems encountered during the ground check-out and the solutions used.

As initially installed, a rigging requirement for spoiler travel required shortening the command position transducer by ½ inch. This shifted the aileron surface null to 7.5 degrees down, which did not meet the required 0 degree null position. The problem was solved by changing the values of two sets of resistors. Figure 23a and 23b, shows one complete control channel (one half the control electronics) of the DDCV system and the resistor sets changed. The first set changed was resistors R1 through R8 which are used to establish the feedback potentiometer gains. As shown in Figure 23a these resistors are in series with the feedback potentiometer resistance elements. Resistors R2, R5, R8, and R11 were changed from the 4.99×10^3 ohm value shown in Figure 23a to 4.11×10^3 ohms. Resistors R3, R6, R9, and R12 were changed to 5.9×10^3 ohms from the 4.99×10^3 ohms value shown in Figure 23a. This null change required changing the set of input resistors R1 of the monitor amplifiers and the command and disconnect amplifiers in order to retain 32 degrees of aileron travel. The input resistors were changed from a value of 100×10^3 ohms as shown in Figure 23a to 61.9×10^3 ohms.

During rigging check-out, the aileron surface oscillated at an amplitude of approximately ± 4 degrees during retract motion. This problem was traced to air being trapped in the oil-filled magnet cavities of the direct drive valve. These cavities are connected through filter discs to the same return pressure. However, the filter discs provide isolation of the two cavities for rapid changes of return line pressure. When the hydraulic oil

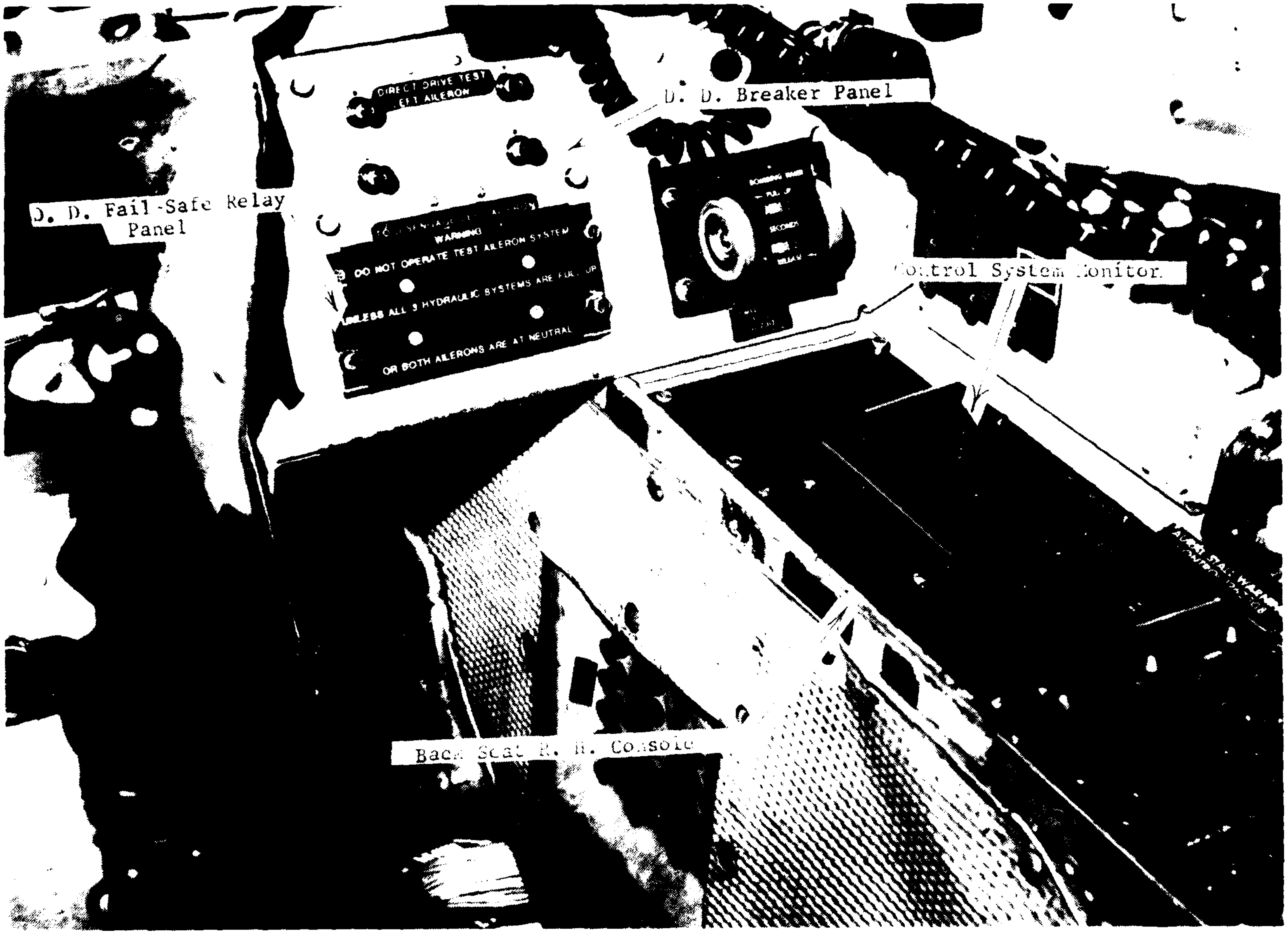


Figure 22 DCI Pilot Monitor Installation

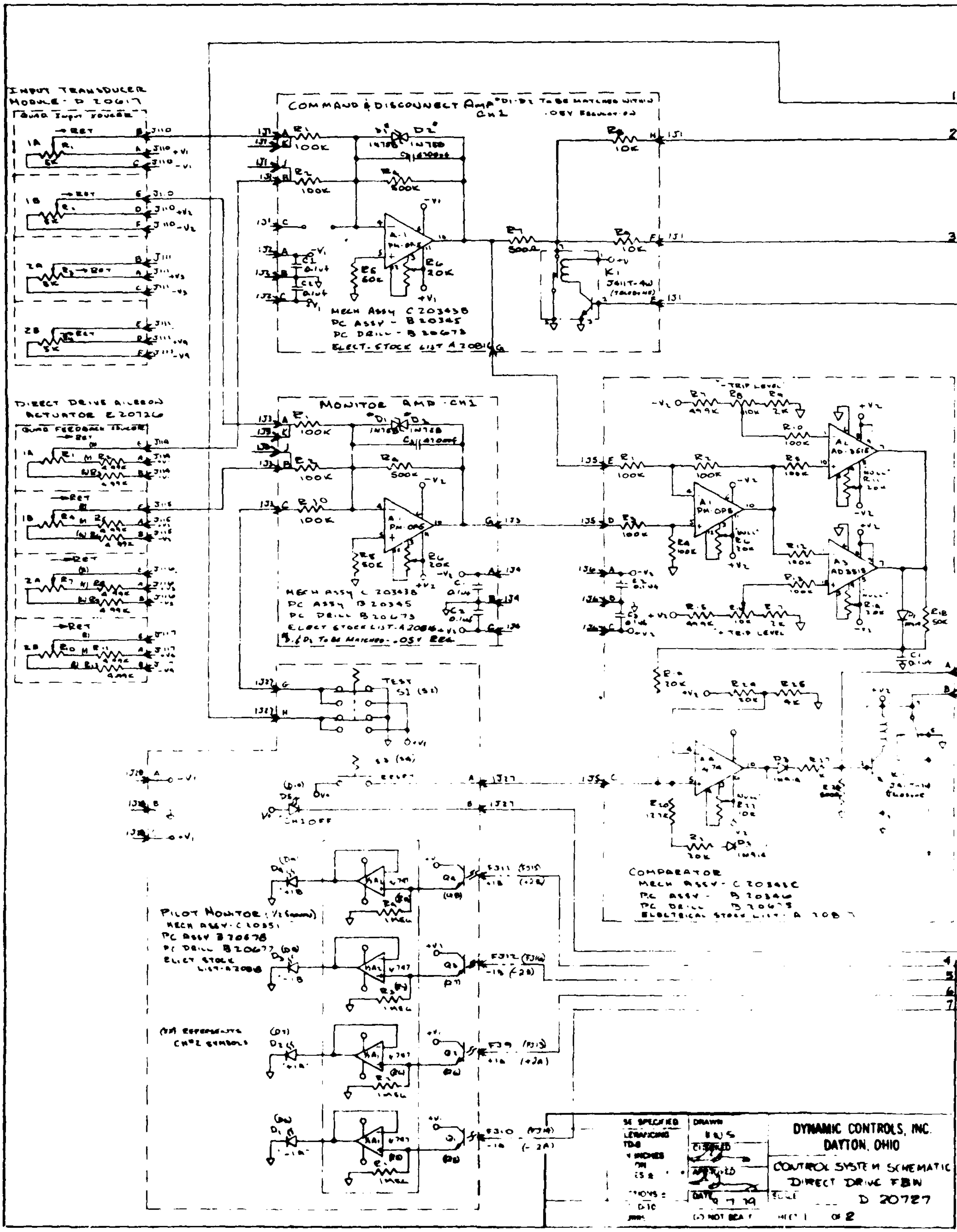
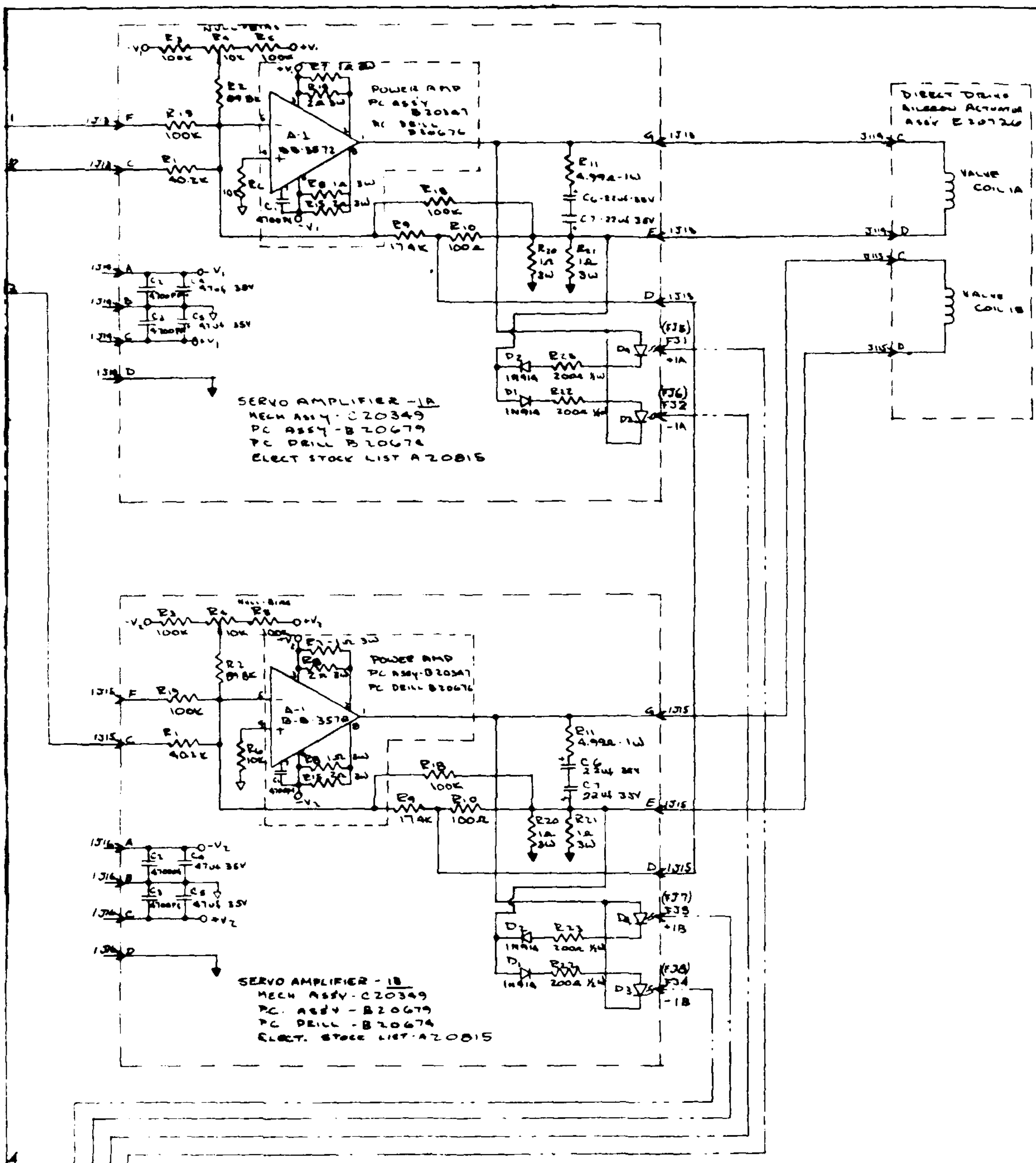


Figure 23a DCI Control System Schematic



NOTE: 1) THIS SCHEMATIC REPRESENTS ½ OR CHANNEL #1 OF A (2) CHANNEL FBW REDUNDANT CONTROL SYSTEM!

CHANNEL #2 CONNECTOR 'J' NUMBERS ARE AS FOLLOWS:

MODULE	SIGNAL	POWER	FIBRE OPTICS +	FIBRE OPTICS −
COMMAND & DISC	1J7	1J8	N/A	N/A
MONITOR	1J9	1J10	N/A	N/A
COMPARATOR	1J11	1J12	N/A	N/A
SERVO AMP 2A	1J17	1J18	FJ5	FJ6
SERVO AMP 2B	1J19	1J20	FJ7	FJ8
PILOT MONITOR	1J29	1J30	FJ13-FJ15	FJ14-FJ16

2) FOR CABLE DIAGRAM SEE D-20728

3) FOR POWER SUPPLY SCHEMATIC SEE C-20825

DIMENSIONS & TOLERANCING PER MIL STD 8
DIMENSIONS ARE IN INCHES
TOLERANCES ON DECIMALS .XX ± .010 .XXX ± .005 ANGLES ± FRACTIONS ±
FINISH PER ... TD-10
REMOVE ALL BURRS

DRAWN HWS
CHECKED
APPROVED
DATE 4-7-79
DO NOT SCALE

DYNAMIC CONTROLS, INC.
DAYTON, OHIO

CONTROL SYSTEM SCHEMATIC DIRECT DRIVE FBW

SCALE | D | 20727
SHEET 2 OF 2

Figure 23b DCI Control System Schematic

in the cavities contains entrained air, rapid changes in return line pressure creates different transient pressures in the magnet cavities. This generates differential pressures across the control spool which causes the retract motion oscillations. Contributing to the problem was the air introduced into the aircraft system by the particular ground carts used during the test program. These carts had no provision for deaerating hydraulic oil. In fact, other control surfaces of the aircraft also oscillated after connection to a ground cart and had to be exercised to clear the air that had been introduced. Also contributing to the problem was the return line pressure surges experienced at the F-4 aileron actuator. The flow restrictions of the particular plumbing of the F-4 wing create large (greater than 1,500 psi) return line pressure transients at the aileron actuator during normal operation of the control surfaces. Interconnecting the magnet cavities directly to each other and then through a single filter disc to return would have eliminated the transient differential pressure problem. This solution would have required a hardware modification which would, because of the time constraints, have been very difficult to implement. The problem was therefore solved by thoroughly bleeding the system and using a hydraulic ground cart having the least amount of entrained air. The bleeding of the system was accomplished by first operating all the control surfaces and then bleeding the aircraft reservoirs. This procedure was repeated until the test system problem cleared up.

During EMI ground check-out (with the engines running and the system operating on aircraft power), Channel 1 started failing when the control stick was moved. This problem was traced to a defective command transducer potentiometer in Channel 1. The problem was solved by replacing the defective potentiometer with a spare.

SECTION V
FLIGHT TESTING

5.1 General

The Air Force Flight Test Center (AFFTC) at Edwards Air Force Base, California assumed the prime responsibility for the flight testing of the two Direct Drive Control Systems submitted for evaluation. Dynamic Controls, Inc. provided on-site support during the flight test planning and the flight tests, as well as during the installation and ground testing. During the flight test, DCI provided pre and post flight functional inspections.

The objective of the flight testing was to evaluate the performance of the two Direct Drive Control Systems. Since the test systems were designed to perform like the normal aileron control system, the flight test program was directed at generating data which would allow comparison of the test system's performance with the normal aileron of the opposite wing.

Each DDCV control system was to be flight tested for 25 hours. The first, thirteenth and twenty-fifth hour of flight tests were designated as data flights. The data flight required performance of a prescribed set of maneuvers as listed in Table 2. This test procedure would allow evaluation of both left and right lateral responses. Each roll maneuver was to be held for 2 seconds or a 360 degree roll, which ever occured first. The input rates shown on Table 2 are defined as follows:

Slow Input - 4 seconds from neutral to full right or left

Medium Input - 2 seconds from neutral to full right or left

High Input - as fast as possible from neutral to full right or left

The data recorded during the data flights was to be reduced by the AFFTC Data Center.

During all test flights not designated as data flights, the instrumentation would be active in order to determine the cause of any inflight problems which could occur.

5.2 Aircraft Instrumentation

For the purpose of recording data for the flight safety testing and the comparison data collection, the aircraft was instrumented to record the parameters listed in Table 3. All data parameters instrumented were measured by analog sensors and then converted to a digital signal for recording on magnetic tape. The sample rate was set at 200 samples per second, providing an adequate sample rate since the maximum dynamic signal of interest was 10 Hz Signal resolution was set at 1024 counts full scale with a maximum of 2 bits uncertainty (providing a signal resolution of .297%). Signal filters of 25 Hz (-3 Db down) were designed into the analog signal conditioners to provide noise rejection for the recorded data.

The primary parameters recorded for the DDCV performance evaluation were lateral stick position, command inputs 1, 2, 3 and 4 (from the input transducer), actuator positions 1, 2, 3 and 4 (left aileron position), right aileron position, roll rate and lateral acceleration. Figure 24 shows the installation of the data recorder in the aircraft nose.

Calibration checks of the data system were performed during preflight checkout and after installation of a new recording tape.

TABLE 2 DDCV SYSTEM PERFORMANCE TESTS

Run No.	Lateral Stick Input	Input Rate	Airspeed (KIAS)	Mach Number	Pressure Altitude (ft)	Comments
1.1	1/4 RT	med	210	0.38	10,000	
1.2	1/4 LT	med	210	0.38	10,000	
1.3	1/2 RT	high	210	0.38	10,000	
1.4	1/2 LT	high	210	0.38	10,000	
1.5	full RT	high	210	0.38	10,000	
1.6	full LT	high	210	0.38	10,000	
2.1	1/2 RT	med	350	0.63	10,000	
2.2	1/2 LT	med	350	0.63	10,000	
2.3	full RT	high	350	0.63	10,000	
2.4	full LT	high	350	0.63	10,000	
3.1	1/2 RT	slow	550	0.98	10,000	
3.2	1/2 LT	slow	550	0.98	10,000	
3.3	full RT	high	550	0.98	10,000	
3.4	full LT	high	550	0.98	10,000	
3.5	full RT	high	550	0.98	10,000	Ch 1 off
3.6	full RT	high	550	0.98	10,000	Ch 2 off
3.7	full LT	med	550	0.98	10,000	doublet
4.1	1/2 RT	med	255	0.63	25,000	
4.2	1/2 LT	med	255	0.63	25,000	
5.1	full RT	high	350	0.82	25,000	
5.2	full LT	high	350	0.82	25,000	
5.3	full RT	high	350	0.82	25,000	Ch 1 off
6.1	full RT	high	550	1.27	25,000	
6.2	full LT	high	550	1.27	25,000	
7.1	1/2 RT	med	250	0.85	40,000	
7.2	1/2 LT	med	250	0.85	40,000	
7.3	1/2 RT	med	250	0.85	40,000	doublet
7.4	full RT	slow	250	0.85	40,000	
7.5	full LT	slow	250	0.85	40,000	
7.6	full RT	slow	250	0.85	40,000	Ch 2 off
8.1	1/2 RT	slow	400	1.25	40,000	
8.2	1/2 LT	slow	400	1.25	40,000	
9.1	1/4 RT	slow	625	1.92	40,000	
9.2	1/4 LT	slow	625	1.92	40,000	
9.3	full RT	high	625	1.92	40,000	
9.4	full LT	high	625	1.92	40,000	
9.5	full RT	high	625	1.92	40,000	Ch 1 off

TABLE 3 PARAMETER LIST FOR DDCV FLY-BY-WIRE ACTUATOR TEST

Parameter #	Description	Range	Freq	Filter
1	Command Input 1	± 10VDC	25Hz	Yes
2	Command Input 2	± 10VDC	25Hz	Yes
3	Command Input 3	± 10VDC	25Hz	Yes
4	Command Input 4	± 10DVC	25Hz	Yes
5	Acuator Pos 1	± 5VDC	25Hz	Yes
6	Acuator Pos 2	± 5VDC	25Hz	Yes
7	Acuator Pos 3	± 5VDC	25Hz	Yes
8	Acuator Pos 4	± 5VDC	25Hz	Yes
9	Roll Rate	± 200 Deg	25Hz	No
10	Pitch Rate	± 60 Deg/Sec	25Hz	No
11	Yaw Rate	± 200 Deg/Sec	25Hz	No
12	Right Aileron Pos	± 30, - 1 Deg	25Hz	No
13	LH Spoiler	0, + 45 Deg	25Hz	No
14	RH Spoiler	0, + 45 Deg	25Hz	No
15	Horiz Stab Pos	+ 8, - 20 Deg	25Hz	No
16	Rudder Pos	± 30 Deg	25Hz	No
17*	KAS	0 - 750 Kts	--	No
18	CG Accel Vert	± 10 G's	25Hz	Yes
19	CG Lat Accel	± 1 G	25Hz	Yes
20	CG Long Accel	± 1 G	25Hz	Yes
21	Lateral Stick Pos	± 8 In	25Hz	No

* Hand Recorded

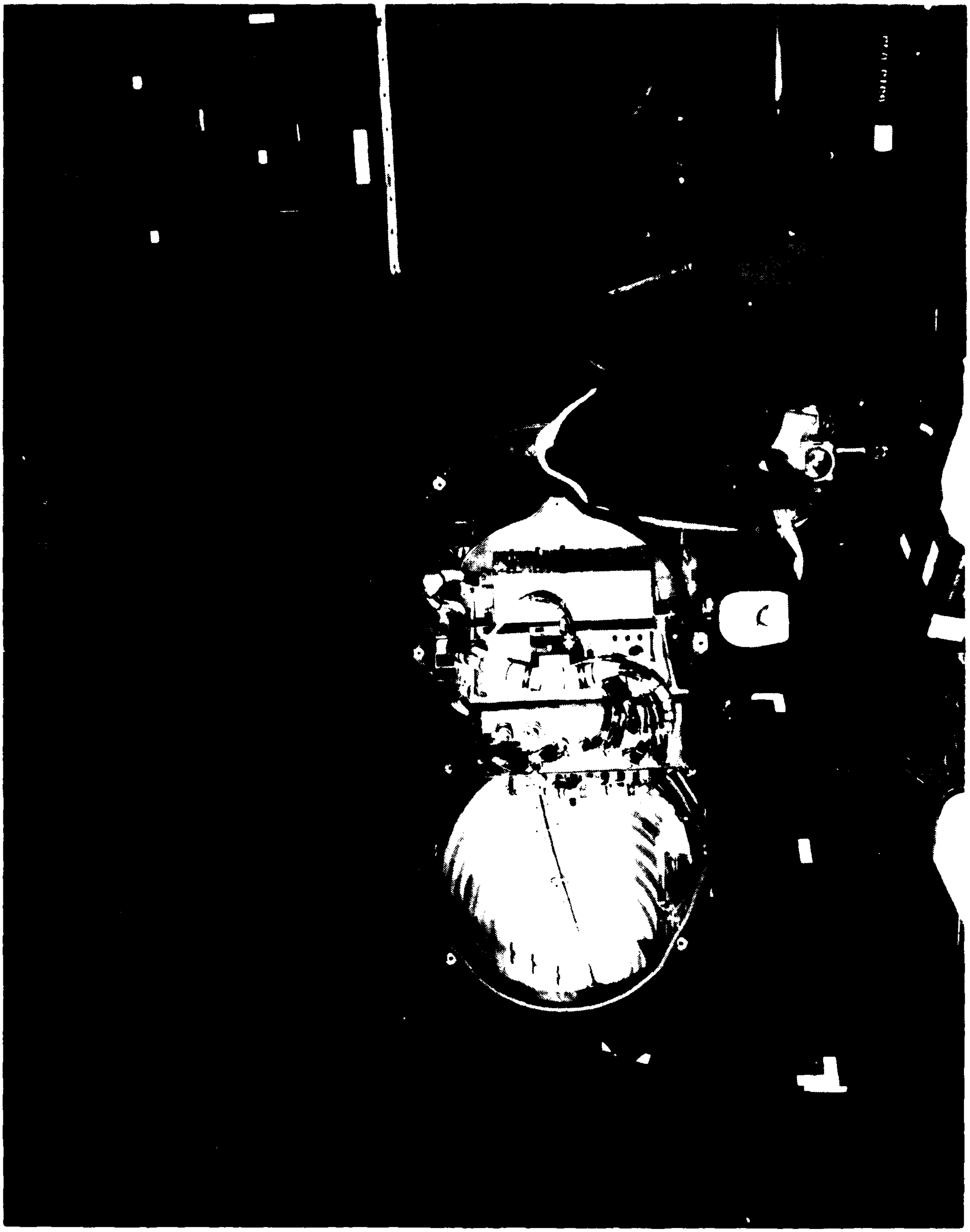
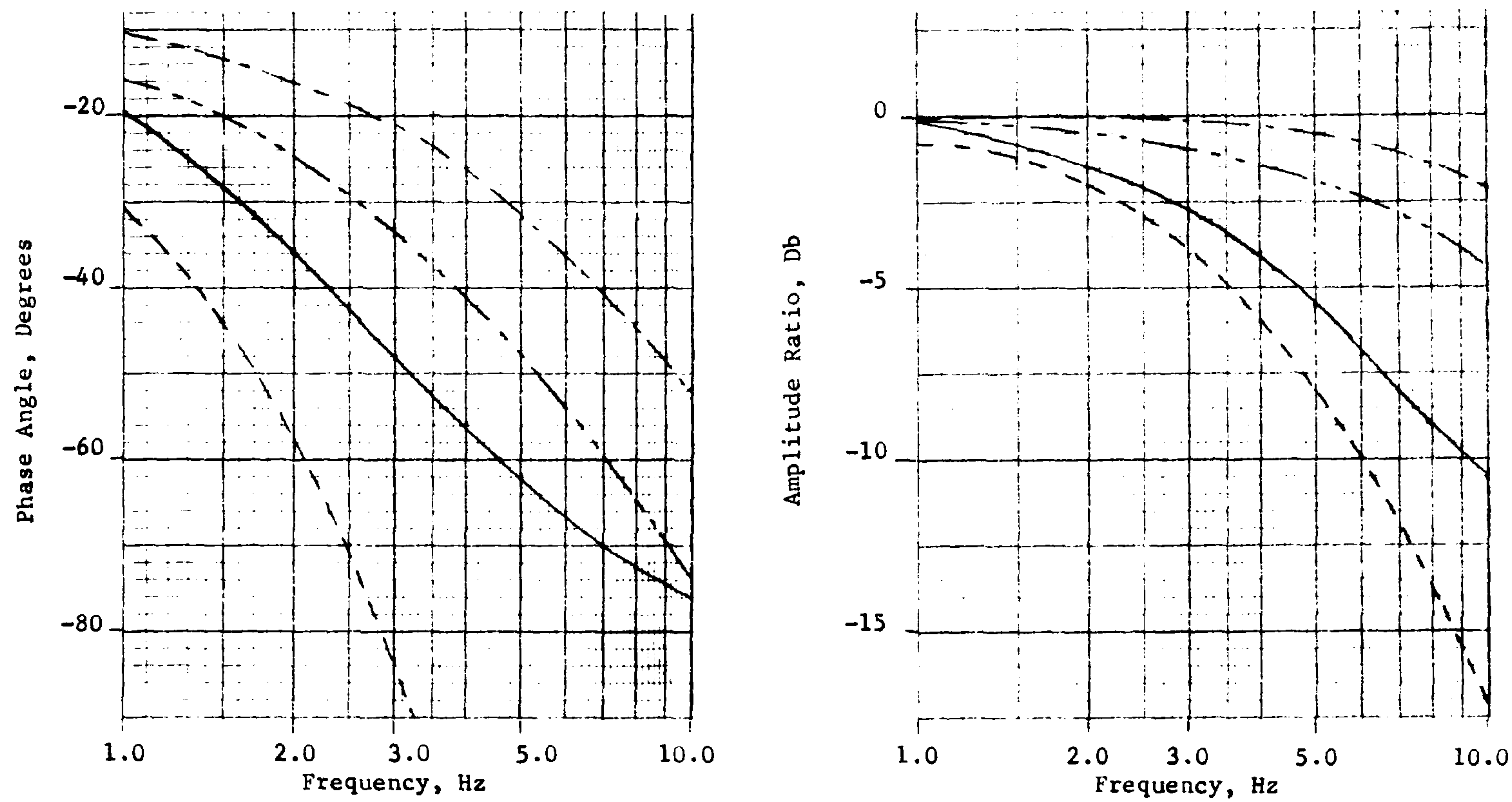


Figure 24. [illegible] installation



Frequency Response *of the General Electric,*
Dynamic Controls, Inc. and Normal F4E Aileron
Power Control Actuator.

Input Amplitude 10% Full Scale.

Key

—— Normal F4E
— · — DCI, 2 Channel
— · · — DCI, 1 Channel
- - - - G. E. System

Figure 25 Systems Frequency Response

5.3 Flight Safety Testing

5.3.1 General

Flight Safety testing consisted of specific performance tests of the two systems both on the ground and during the initial flights. The following section describes the purpose of the tests and the procedure used.

5.3.2 Aircraft System Stability

As a result of an AFFTC review of both DDCV systems for their effect upon aircraft stability, it was concluded that both systems required special flight safety testing procedures. This was because the frequency response characteristics of both DDCV systems were not an exact match for the normal F-4 aileron actuator. Figure 25 shows the frequency response measured by Dynamic Controls, Inc. on both the DCI and the G. E. systems. Also shown in Figure 25 is the frequency response of the normal aileron actuator as experimentally determined by McDonnell Aircraft Company (and presented in AFFDL-TR-72-116, "Active Flutter Suppression *Systems for Military Aircraft* A Feasibility Study" on page 194).

5.3.2.1 G. E. DDCV Stability Considerations

The G. E. system created some concern with respect to a potential limit cycle with the lateral axis stability augmentation system engaged. This was because the G. E. DDCV system exhibited considerably greater phase shift with increasing frequency than the normal aileron actuator.

To verify flight safety of operation, a flight test procedure was planned in which the aircraft speed was increased with the stability augmentation turned on. An observer aircraft would monitor the left aileron surface for visible limit cycle motion while the pilot performed "stick rap" maneuvers.

The expected characteristic of the potential limit cycle was that the amplitude would start at a low value and increase with increasing airspeed. Engagement of the lateral axis stability augmentation system would not be made until the aircraft had obtained an altitude of 10,000 feet. If a limit cycle was observed, the stability augmentation system would be turned off and the aircraft returned to base.

5.3.2.2 DCI DDCV Stability Considerations

The DCI DDCV system was designed to have approximately twice the frequency response of the normal aileron actuator system with both channels of the DDCV control system operating. With one channel failed, the frequency response changed to one half that of two channel operation. This characteristic was associated with the design philosophy of not using gain changing with failures or force motor position feedback in order to have the minimum number of components in the mechanization. Figure 25 shows the DCI system measured responses of both the 2 channel and 1 channel operation. With both operating conditions, the amplitude degradation and phase lag were less than that of the normal F-4 aileron system.

The major concern with the DCI DDCV system was one of exciting structural resonance modes. To investigate that possibility, two test procedures were planned. The first procedure was a ground test which involved deflating the tires to one half normal pressure in order to isolate the structure from the ground. The control system with the stability augmentation system turned on was excited by sharp "stick raps". Structural mode excitation would be indicated by sustained oscillations of the aileron surface caused by the stability augmentation system feeding back a structural mode. If the ground test was passed successfully, a second test procedure for an inflight limit cycle would be conducted. This consisted of using sharp "stick raps" as an input to excite the system during flight. The aircraft would be flown at an altitude of 10,000 feet with a chase aircraft monitoring the test aircraft for sustained surface oscillations occurring after a "roll right" input. The test sequence would start at Mach 0.5 with the stability augmentation engaged. Airspeed increases of Mach .05 would be made after a successful completion of each stick rap test. The airspeed would be increased through Mach .90, in order to obtain the maximum dynamic pressure condition. Should any sustained oscillations be observed, the stability augmentation system would be turned off and the aircraft returned to base.

5.3.3 Ground Check Testing

The DDCV systems required a ground check-out after installation in order to verify their proper operation. The following test sequence was established to evaluate the system using ground power:

1. Perform the test system's pilot preflight check list (as listed in the Partial Flight Manual for each test system shown on Figure 26 for the G. E. system and Figure 27 for the DCI system).

2. Operate the test system on PC 1 and utility hydraulic system individually and perform the preflight check list.

3. Record for comparison on X-Y plots: stick position-vs-right aileron position, stick position-vs-left aileron position, stick position-vs-input transducer position and input tranducer position-vs-left aileron position.

4. Check the test aileron surface for transients and final null offset encountered upon a single channel failure.

5. Record the comparison left and right aileron maximum velocities.

6. Time drift rate from full down to full up when both channels are failed. (The time should be within the range of 6 to 15 seconds).

NORMAL PROCEDURES

NOTE

These procedures apply to the General Electric System.

<u>AFTER ELECTRICAL POWER</u>

1. (WSO) Four test circuit breakers-PUSH-IN.

<u>BEFORE TAXIING</u>

NOTE

These procedures are to be accomplished prior to the normal F-4E before taxiing (Front Cockpit) preflight check.

1. (WSO)* Channel 1 and 2 - RESET
2. (WSO) Channel 1 "LVDT" switch - ACTUATE
3. (WSO) "CH 1 OFF" light - CHECK ON
4. (WSO) Channel 1 - RESET
5. (WSO) Channel 1 "SERVO" switch - ACTUATE
   (P)** Control stick - MOVE QUICKLY FULL RIGHT
6. (WSO) "CH 1 OFF" light - CHECK ON
7. (WSO) Channel 1 - RESET
8. (WSO) Channel 1 "COMM" switch - ACTUATE
9. (WSO) "CH 1 OFF" light - CHECK ON
10. (WSO) Channel 1 - RESET
11. (WSO) Channel 2 "LVDT" switch - ACTUATE
12. (WSO) " CH 2 OFF" light - CHECK ON
13. (WSO) Channel 2 RESET

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14. (WSO) Channel 2 "SERVO" switch ACTUATE
    (P) Control stick - MOVE QUICKLY FULL RIGHT
15. (WSO) "CH 2 OFF" light - CHECK ON
16. (WSO) Channel 2 - RESET
17. (WSO) Channel 2 "COMM" switch - ACTUATE
18. (WSO) "CH 2 OFF" light - CHECK ON
19. (WSO) Channel 2 - RESET
20. (P) Control Stick - HOLD FULL RIGHT
21. (WSO) Channel 1 and 2 "COMM" switches - ACTIVATE
22. (WSO) Left aileron - CHECK FULL UP WITHIN 15 SEC.
23. (P) Control stick - CENTER
24. (WSO) Channels 1 and 2 - RESET

WARNING

(WSO) Reset channels 1 and 2 after all checks in the normal F-4E checklist which call for depressing the emergency quick release.

* (WSO) - WEAPONS SYSTEM OFFICER (Rear Seat)

** (P) - PILOT (Front Seat)

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Figure 26 Pilot's Preflight Checklist - G.E. System

NORMAL PROCEDURES

NOTE

These procedures apply to the Dynamic Controls, Inc. System.

AFTER ELECTRICAL POWER

1. (WSO) Four test circuit breakers - PUSH IN
2. (WSO) Channel 1 and 2 - RESET
3. (P) Control stick - MOVE TO LEFT STOP
4. (WSO) Channel 1A, 1B, 1A and 2B "+" lights - CHECK ON
5. (P) Control stick - MOVE TO NEUTRAL
6. (WSO) Channel 1A, 1B, 2A and 2B "+" lights - CHECK OFF
7. (P) Control stick - MOVE TO RIGHT STOP
8. (WSO) Channel 1A, 1B, 2A and 2B "-" lights - CHECK ON
9. (P) Control stick - MOVE TO NEUTRAL
10. (WSO) Channel 1A, 1B, 2A and 2B "-" lights - CHECK OFF

BEFORE TAXIING

NOTE

These procedures are to be accomplished prior to the normal F-4E before taxiing (Front Cockpit) preflight check.

1. (P) Control stick - CENTER
2. (WSO) Channel 1 "TEST" switch - ACTUATE
3. (WSO) "CH 1 OFF", channel 1A "-" and channel 1B "+" lights - CHECK ON
4. (WSO) Channel 1 - RESET

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5. (P) Control stick - CENTER
6. (WSO) Channel 2 "TEST" switch - ACTUATE
7. (WSO) "CH2 OFF", channel 2A "-" and channel 2B "+" lights - CHECK ON
8. (WSO) Channel 2 - RESET
9. (P) Control stick - MOVE FULL RIGHT
10. (WSO) Channel 1 and 2 "TEST" switches - ACTUATE
11. (WSO) Left aileron - CHECK FULL UP
12. (P) Control stick - CENTER WITHIN 15 SECONDS
13. (WSO) Channels 1 and 2 - RESET

WARNING

(WSO) Reset channels 1 and 2 after all checks in the normal F-4E checklist which call for depressing the emergency quick release lever.

* (WSO) - WEAPONS SYSTEM OFFICER (Rear Seat)

** (P) - PILOT (Front Seat)

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Figure 27 Pilot's Preflight Checklist DCI System

The following sequence was to be performed with the engines operating and the test systems connected to internal power.

1. Perform the test system's pilot preflight check list.

2. Perform the EMI testing designated for ground operation.

In addition, after passing the preceeding tests, each system was required to operate satisfactorily on ground power for 10 hours before the aircraft would be released for flight testing.

5.3.4 Electromagnetic Interference Testing

Both the G. E. and DCI system's interface hardware was not tested for electromagnetic interference (EMI) problems prior to the aircraft installation. Although the G. E. system hardware had been flightworthiness tested for EMI during development, the DCI system had been exempted from EMI testing. Therefore, EMI testing was performed as part of the installation check-out of both systems. (Since the signal levels for the systems were not considered low level and the frequency response of the systems limited to 20 Hz, EMI was not anticipated to be a problem with the systems and the testing would be done upon aircraft installation.)

The EMI testing was conducted in three phases. The first phase was with ground power. The second phase was with the aircraft on the ground and operating under aircraft power. The third phase was during flight (in order to check the effect of landing gear operation, UHF, TACAN, IFF and the radar altimeter). The general test procedure for aircraft on the ground was a sequential operational check of aircraft switches and equipment, during which the aircraft is left aileron position was monitored for changes and an instrumentation support unit was used to record electrical fluctuations within the test system. Appendix I lists the ground check test procedure used for the two systems.

5.3.5 Flight Test Check-Out Testing

As an operational procedure, a Functional Check Flight (FCF) is required by the Air Force after an aircraft's flight control system is changed or repaired. Therefore, the test aircraft required an FCF after the ground check-out had been satisfactorily completed. The functional check flight procedure was modified for both systems as follows:

1. The aircraft would make a normal takeoff and non-afterburner climb to 10,000 feet altitude above sea level with the gear down. The stability augmentation system for the lateral axis would not be turned on. Upon reaching 10,000 feet and an indicated airspeed of 240 knots, the inflight EMI test of raising the landing gear, cycling the internal and external lights, cycling the pilot heat and operating the identification system (IFF) would be conducted. An observer in a chase aircraft would call out any observed transients and the test aircraft instrumentation would record all events.

2. After the EMI test is satisfactorily completed, a rigging and stability augmentation test (without engaging lateral stability augmentation) would be conducted according to the standard procedure for the F-4 aircraft.

3. Upon satisfactorily completing the rigging check, a check of the aircraft lateral response with the test system disconnected would be run. The procedure used is described in Appendix II of this report. (This test verifies adequate lateral control in the event that the test system is not operational.)

4. If adequate control is exhibited with the test system turned off, then the test system would be reset and the functional flight test according to the pilot's standard FCF check list for the F-4 aircraft would be completed with the lateral augmentation remaining turned off.

5. Upon satisfactory completion of the FCF, the lateral augmentation system would be turned on and the limit cycle/structural resonance tests as described in section 5.3.2 of this report would be conducted.

Upon the successful completion of the preceding tests, the aircraft would be allowed to perform the DDCV prescribed test points (reference Table 2). The aircraft would be released for support flying for all missions except close photo chase, high angle of attack chase and very high speed chase.

5.4 G. E. System Test Results

5.4.1 Ground Test Results

For the ground tests, a mobile data reduction unit was connected to the aircraft instrumentation through the instrument output connector. This unit and an X-Y plotter allowed direct recording of the input-output parameters of the test system.

Figure 28 is an X-Y plot of the right aileron position vs the lateral stick position. This plot is representative of the normal F4E lateral control input-output gain. The hysteresis is 3.0% with an average position gain of 2.48 degrees of aileron travel per degree of lateral stick motion. The stepping in the recorded trace is due to the signal resolution limitations on the digital data acquisition system. The dead band shown on Figure 28 at stick full is built into the control system in order to provide a low null sensitivity in the lateral control.

Figure 29 is an X-Y plot of the DDCV command transducer position vs the lateral stick position. The hysteresis is approximately 3%, is due to play in the normal aircraft linkage and the .7 lb breakout friction level of the command transducer.

Figure 30 is an X-Y plot of the left aileron position vs lateral stick position. This plot shows a position gain of 2.42 degrees aileron deflection per degree lateral stick input. The 3% hysteresis reflects the

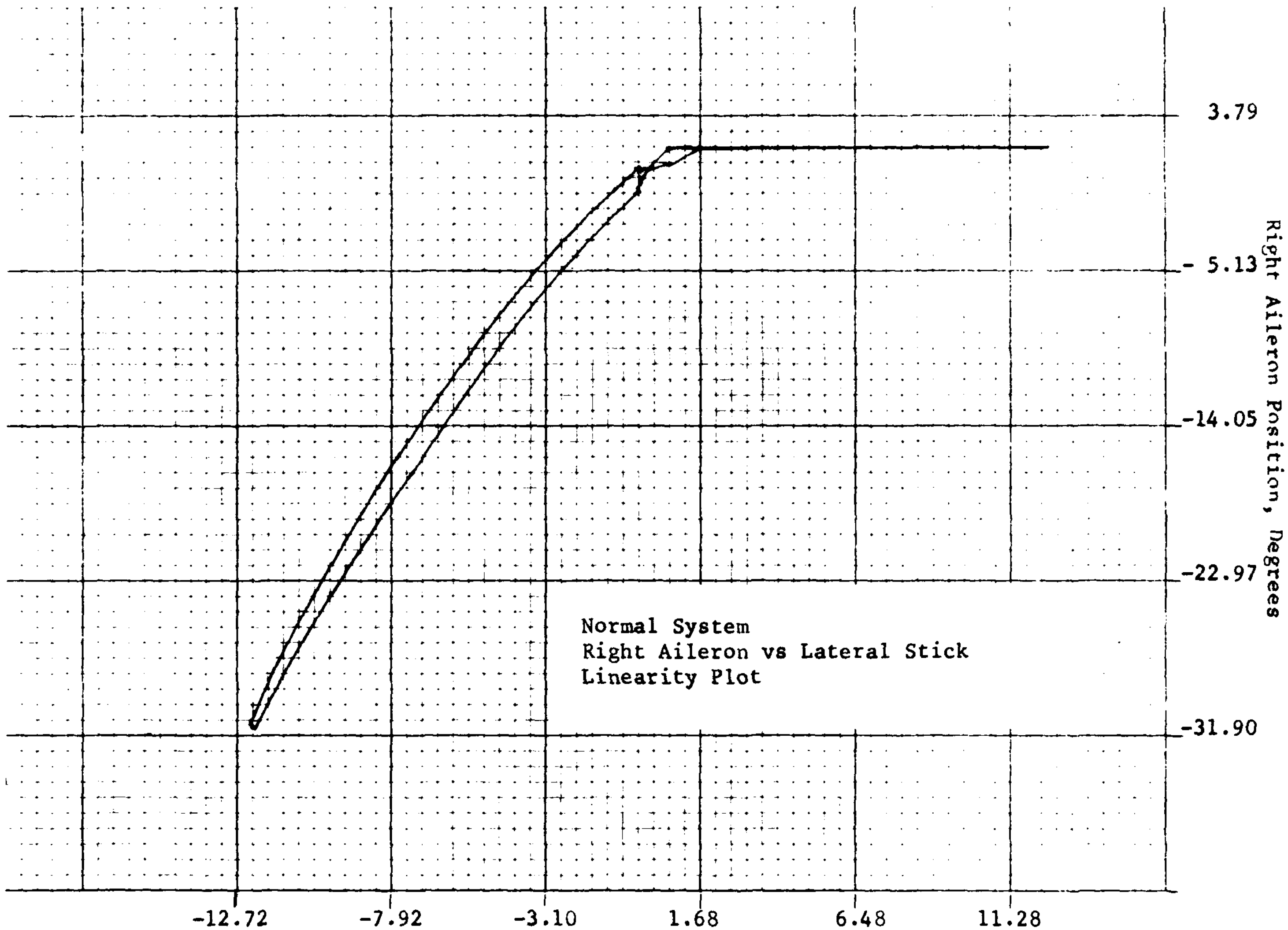


Figure 28 Right Aileron vs Lateral Stick

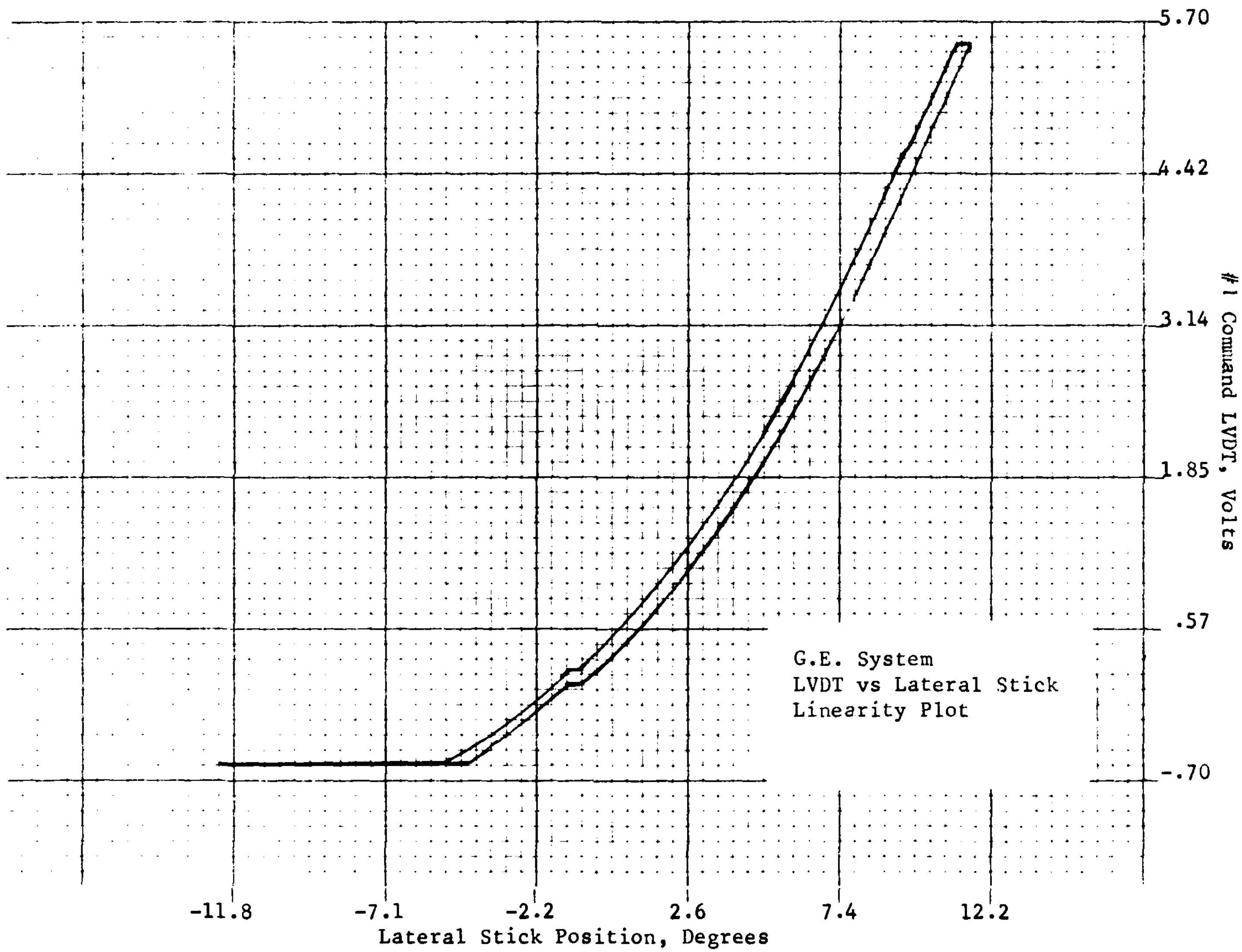


Figure 29 LVDT vs Lateral Stick

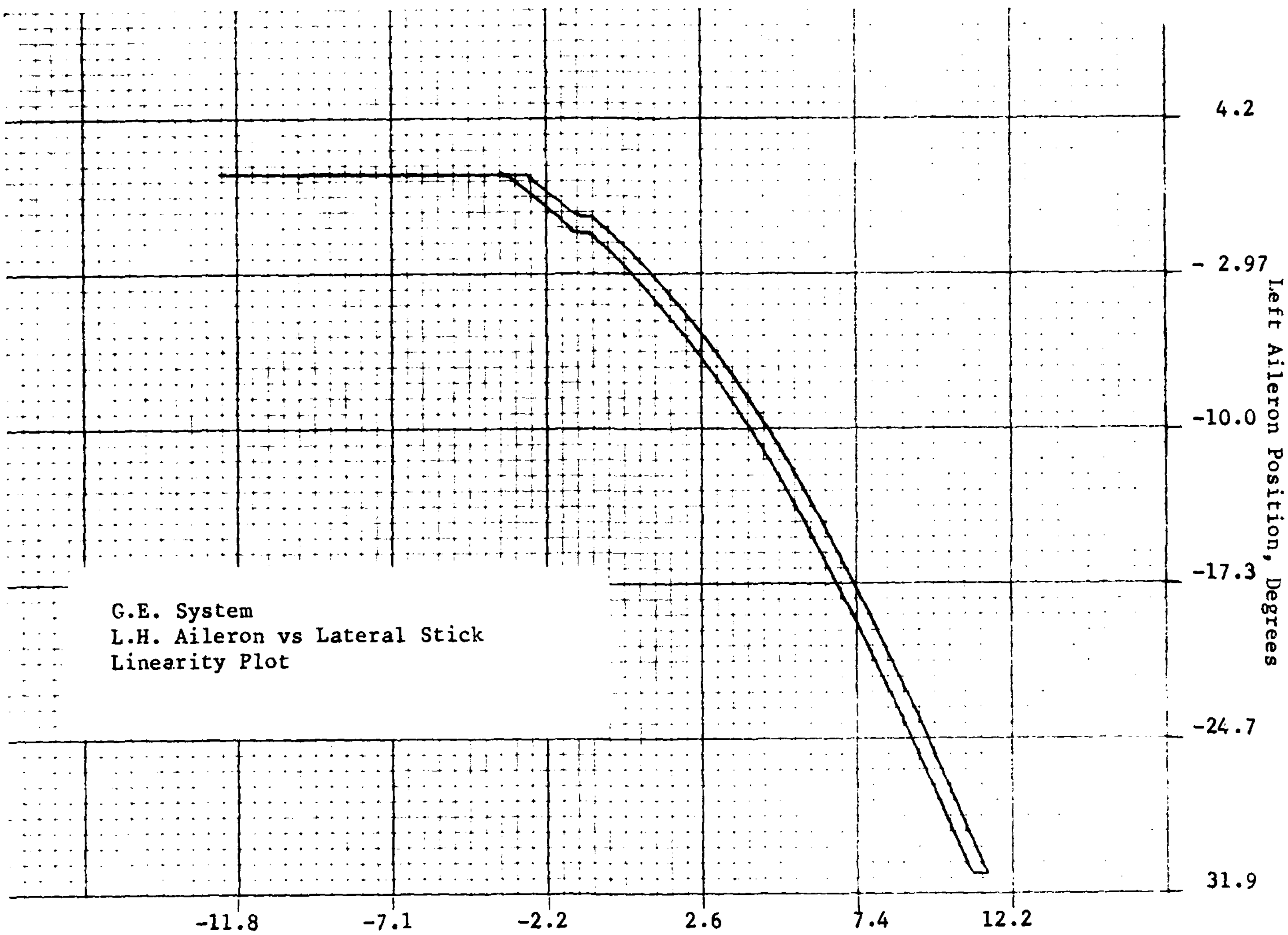


Figure 30 L.H. Aileron vs Lateral Stick

the input linkage and command transducer breakout friction level. Note that there is no significant increase in the hysteresis caused by the addition of the DDCV actuator.

Figure 31 shows the relationship between the lateral stick position and the left and right aileron positions with both channels of the G. E. system operating. The lateral stick was exercised from center to full left and full right positions at the maximum rate that could be applied by hand. Note the stick position changes lead the aileron motions, ensuring that the actuator is moving at maximum rate. During the left stick inputs, the mechanical input linkage stops could be detected during the input motions. As shown on Figure 31, the maximum left aileron velocity is 83.8 degrees per second.

The time drift from full down to full up position of the left aileron upon a 2 channel failure varied from 9 to 12 seconds, depending upon the oil temperature. This was within the 6 to 15 seconds range required for the test system.

The ground EMI tests were successfully completed with the engines running. Both the visual observation and review of the recorded data verified that EMI transients did not exist with the G. E. DDCV system.

5.4.2 Flight Test Results

5.4.2.1 Test Flight No. 1

The mission of this test was to perform the FCF and EMI test checks. The duration of the flight was .3 hours. The preflight and takeoff were normal. The mission was aborted when the left main landing gear would not fully retract. The pilot indicated that the handling in the roll axis felt normal.

5.4.2.2 Test Flight No. 2

The mission of this test flight was to perform the FCF, EMI, lateral control evaluation and limit cycle tests. The duration of the flight was .4 hours. The technical observer in the chase plane did not notice any transients during the inflight EMI testing. The lateral control evaluation was completed with the DDCV test system in the fail safe mode (left aileron full up at +1 degree). According to the pilot, good control of the aircraft was obtained at the 300 knots indicated airspeed (KIAS) cruise mode. Four to five "clicks" of roll right trim were required to maintain level flight. Control above 19.2 units angle of attack (AOA) was minimal. The test pilot felt that the aircraft could land safely with the test system turned off as long as the approach AOA was at or below 17 units. Completion of the test flight mission was aborted due to a boundary layer control light illuminating. (The test aircraft did not have the boundary layer control system. The light should have been labeled "Slat Lockout".)

5.4.2.3 Test Flight No. 3

The mission of this flight was to complete the functional flight test and limit cycle tests. The duration of the test flight was 1.5 hours. The FCF was successfully completed with the lateral augmentation turned off. The limit cycle envelope expansion tests were conducted at an altitude of 10,000 feet at Mach number of .50, .60, .65, .70, .80, .85, .90, and .95. The technical observer in the chase aircraft did not note any sustained oscillations of the left aileron control surface. The test points were limited to a maximum speed of Mach .95 because the chase aircraft was carrying three external fuel tanks and could not follow above that Mach number.

Review of the data from the instrumentation did not show any problems with the limit cycle at the test points. The data review did indicate a noisy roll rate signal and no record of lateral stick position changes. The roll rate gyro mounting fixture was subsequently stiffened and a wire connecting the lateral stick position transducer to instrumentation repaired.

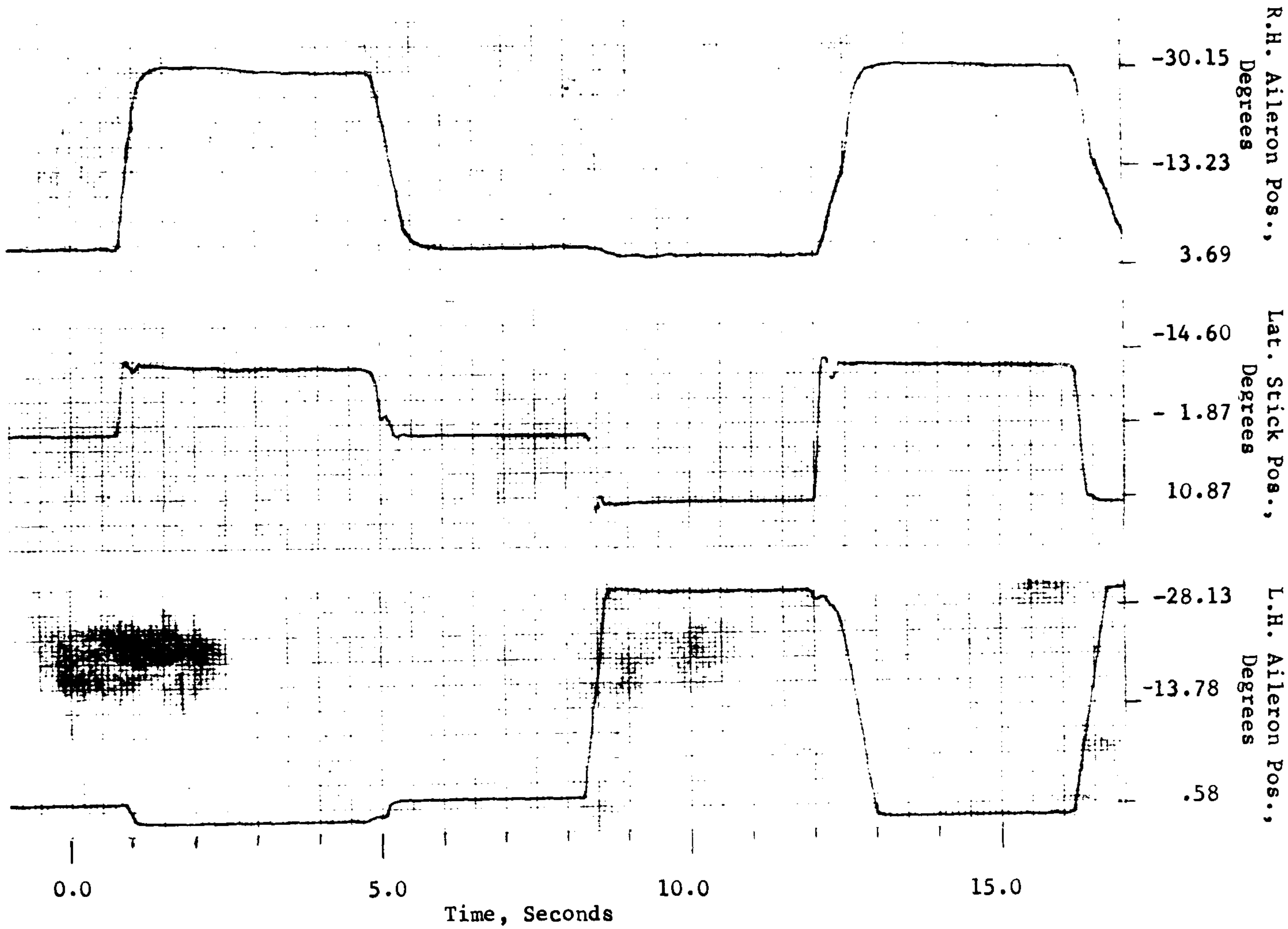


Figure 31 G. E. 2 Channel Velocity Time-History

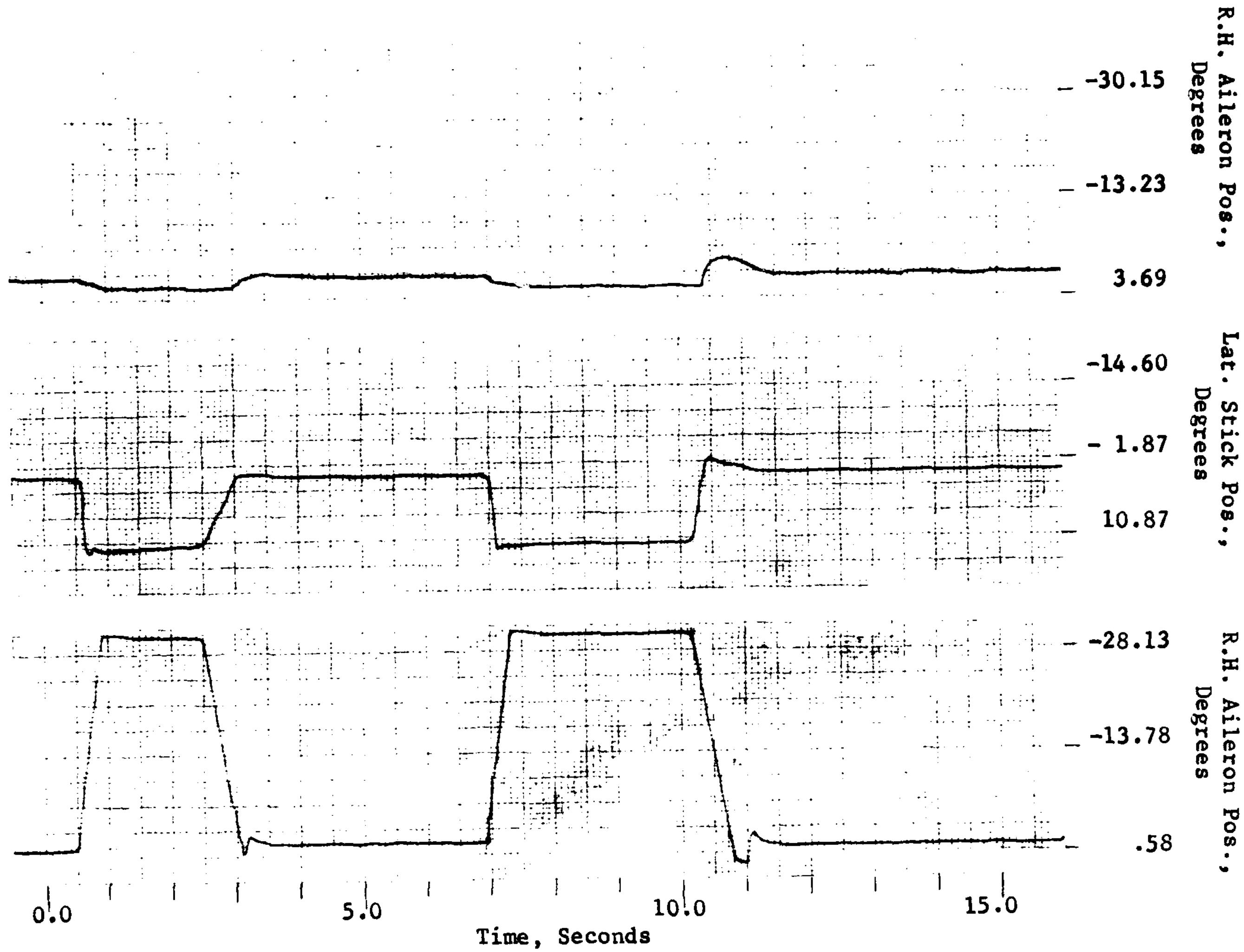


Figure 32 G. E. 1 Channel Velocity Time-History

5.4.2.4 Test Flight No. 4

The mission of this test flight was completion of the limit cycle testing and taking the first set of DDCV performance data. However, a two channel failure was experienced during takeoff. The pilot noted a flicker of the master caution light between rotation and lift-off. Control was maintained (with the test system left off) while accelerating to 10,000 feet, the DDCV system was reset and the aircraft returned to base with no further problems. As a result of the failure, a Safety Board investigation was initiated.

5.4.2.5 Test Flight 4 Two Channel Failure Analysis

The analysis of the G. E. DDCV test systems recorded data indicated that during takeoff, all power to the system electronics was lost for a period of 115 milliseconds. This was established by comparing the channel 1A and 2A actuator position signals as shown on Figure 33. The loss of power is indicated when all position signals went from a negative .20 volts to zero and then returned to the negative .16 volt level. A similar voltage change occurred on the command transducers output signal data recorded simultaneously. Even though power was restored to both channels, the control channels remained in a failed condition. This was because the failure logic in the G. E. computer was designed to detect power losses greater than 40 milliseconds and latch, disconnecting the servo drivers.

In installing the test system, an emergency disconnect system using a single relay was used to control two independent power sources supplied to the two electronics control channels. Figure 34 shows the electrical schematic of the emergency disconnect system. The disconnect switches controlling the disconnect relay were the autopilot and augmentation system paddle switches, mounted on the control stick just below the grip.

The investigation conducted to determine the source of power interrupt was unsuccessful in isolating any single component that caused the power inter-

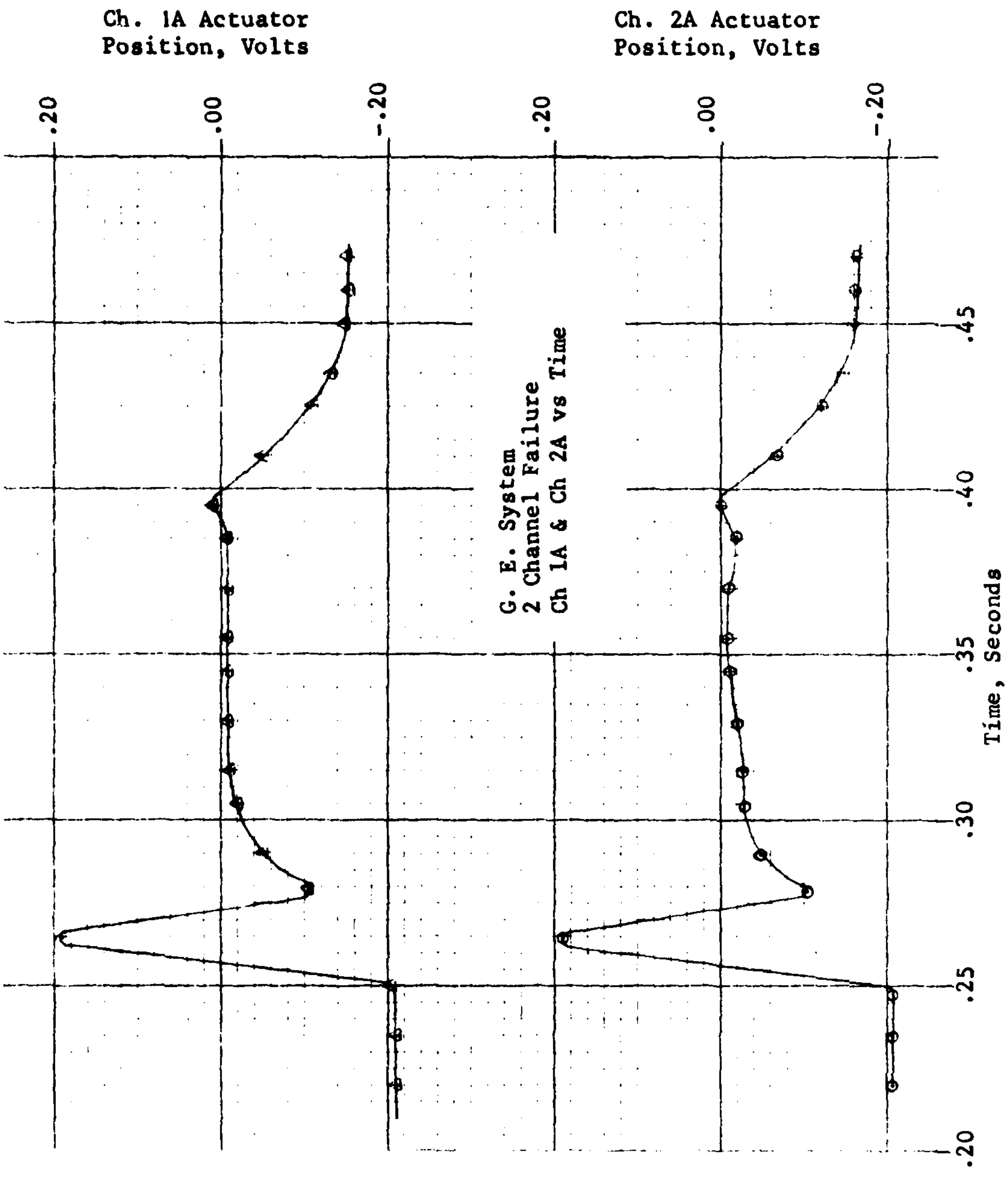


Figure 33 2 Channel Failure

rupt. To avoid further problems, the emergency disconnect system was redesigned to eliminate the single point failure mode by using a separate relay and control for supplying power for each channel of the electronics. Figure 35 illustrates the changed design. In modifying the disconnect system, the Safety Review Board felt that the pilot's two detent gun switch was adequate for use as the test system disconnect device. This approach eliminated a specific switch disconnect for the test engineer in the back seat. The test engineer could disconnect the DDCV system by pulling the power circuit breakers for the test system.

5.4.3 Flight Test Results Summary - G. E. DDCV System

Due to the limited time remaining for the aircraft availability, with the advent of the disconnect problem and the investigation, it was decided that no further flight test time was available for the G. E. DDCV system. Therefore no specific data point measurements were made to permit exact analysis of the relative performance of the DDCV system compared to the normal F-4 aileron system. However, the pilot comments indicated that the test system flew comparably to the normal aircraft lateral control system. The flight time accumulated on the G. E. DDCV system was 2.4 hours over four separate flights.

5.5 DCI System Test Results

5.5.1 Ground Testing

Figure 36 is an X-Y plot of the left aileron position vs the lateral stick position. Compared to the X-Y plot taken on the right aileron shown on Figure 28, the hysteresis is 0.7 degree compared to the 1.6 degree observed on Figure 28. The static gain is identical to the right aileron's at 2.45 degrees aileron deflection per degree lateral stick deflection.

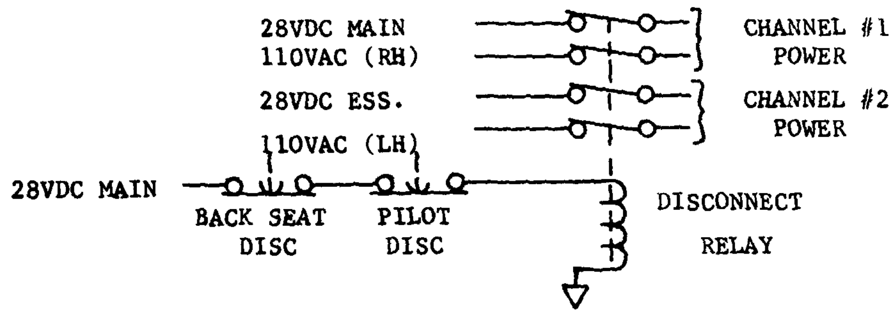


Figure 34 Original Power Disconnect

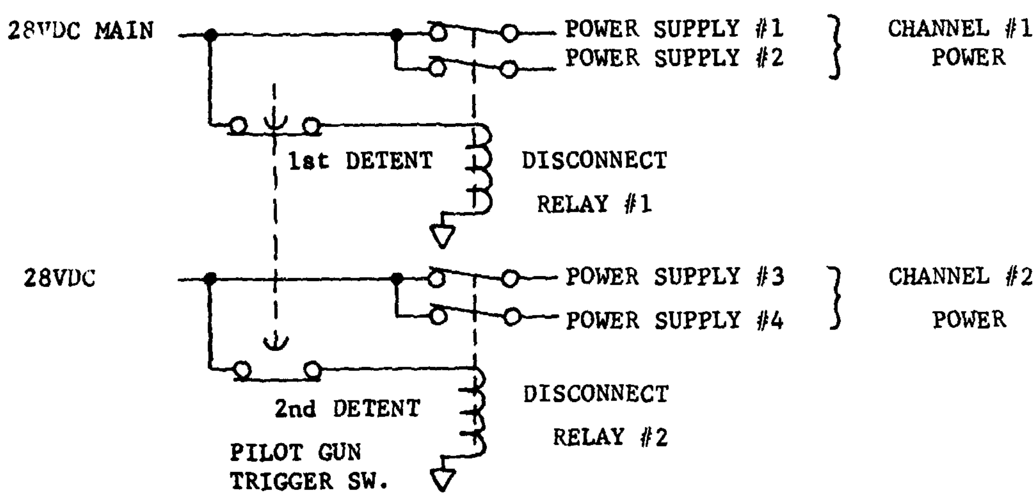


Figure 35 Modified Power Disconnect

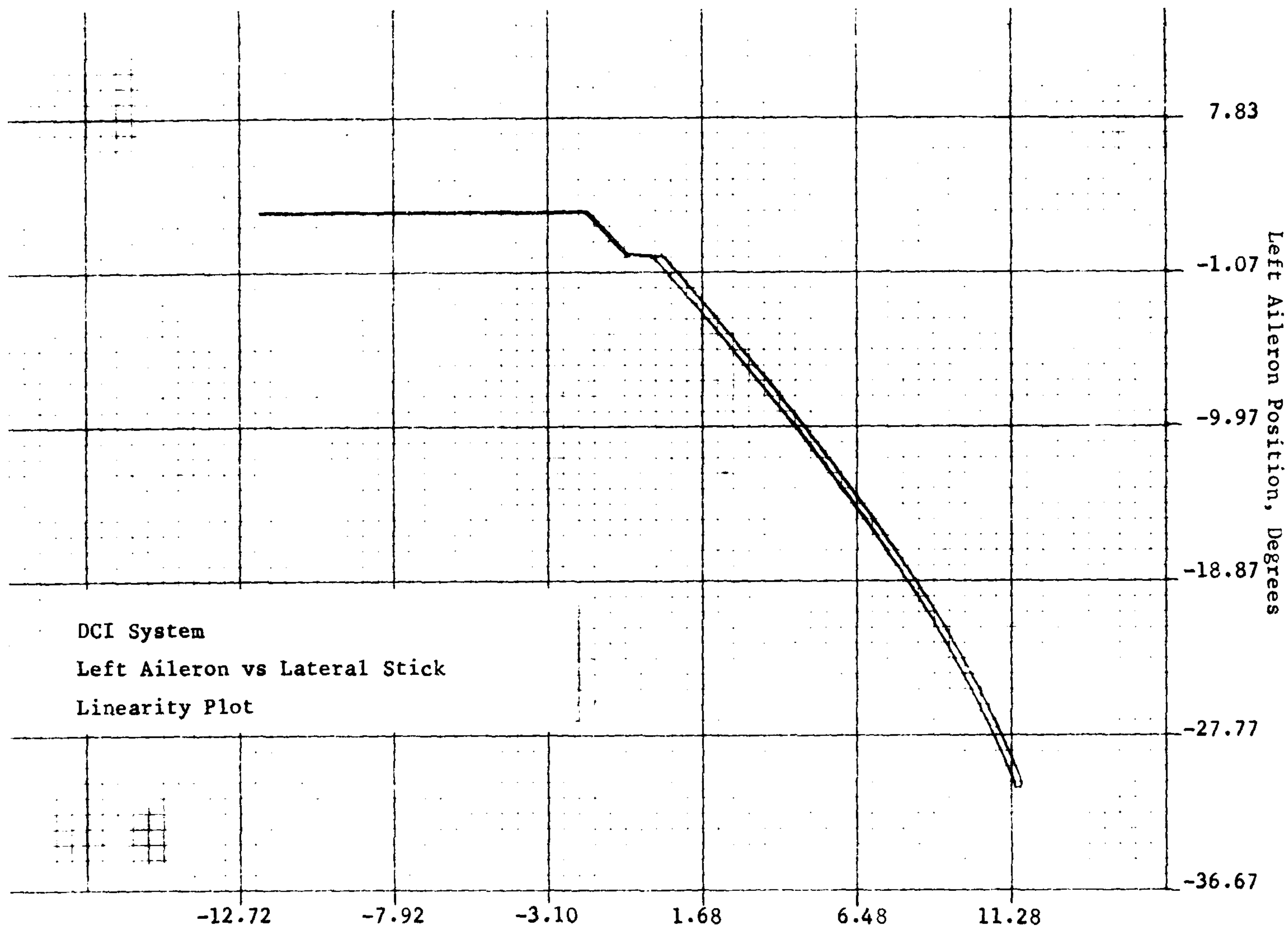


Figure 36 Left Aileron vs Lateral Stick

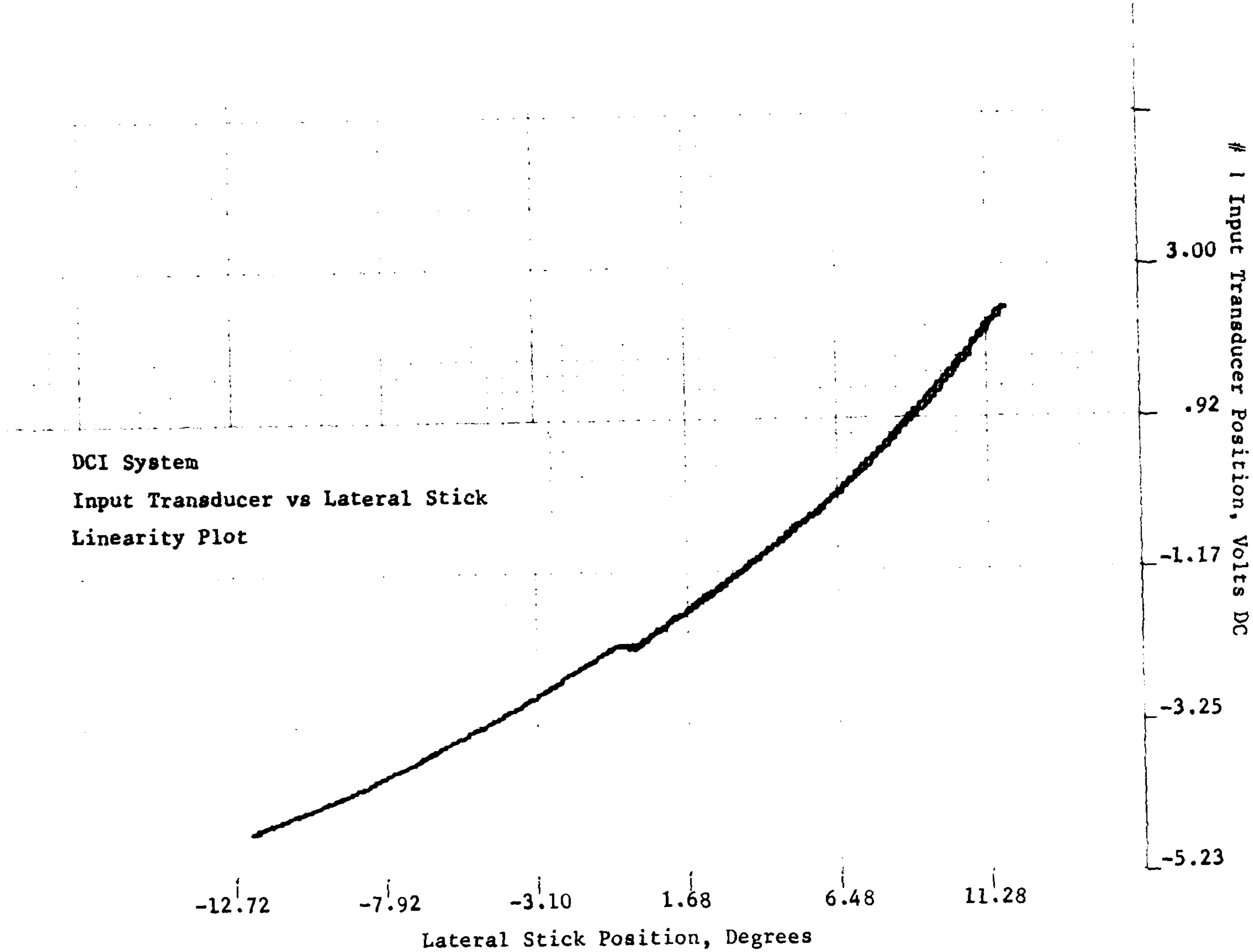


Figure 37 Input Transducer vs Lateral Stick

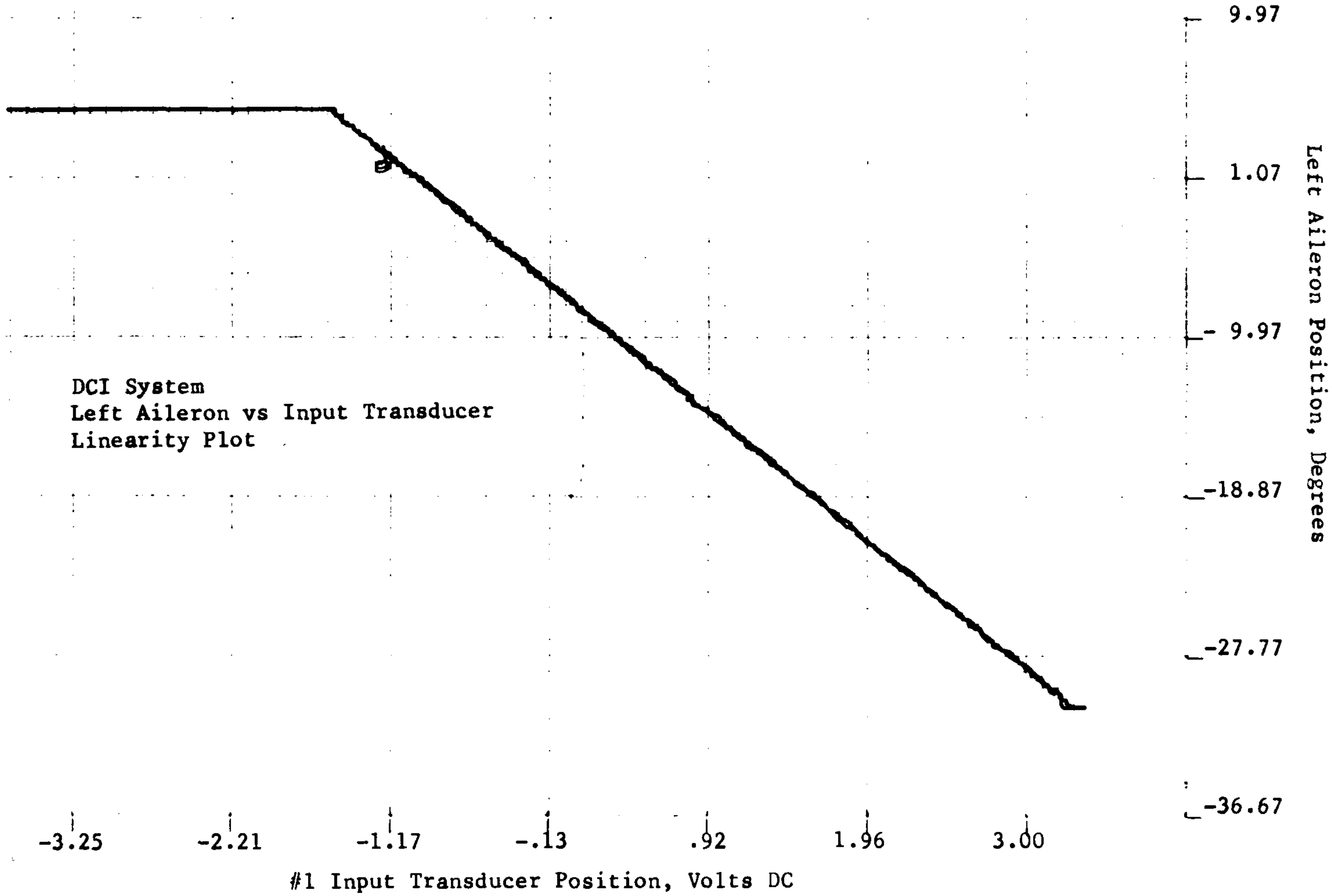


Figure 38 Left Aileron vs Input Transducer

Figure 37 shows the input transducer deflection vs the lateral stick position for section 1A of the input transducer. The non-linearity shown on Figure 37 is designed into the normal input linkage of the aircraft. The average gain is .39 volts/degree of stick input. As shown on Figure 37, the hysteresis is less than 1%. Figure 38 shows the left aileron position vs the input transducer position. This plot is identical to that recorded during laboratory testing.

The maximum velocity of the left aileron was measured at 88 degrees per second for downward motion and 55 degrees per second for aileron up motion. This compared satisfactorily to the 83 degrees per second down and 55 degrees per second up velocities for the standard aileron surface.

The EMI ground check list revealed no surface position disturbances due to EMI generated by the aircraft systems. The ground structural resonance coupling testing was successfully completed with no regenerative structural mode excitation excited by "stick raps" with the roll augmentation system engaged.

The fail safe drift rate from the aileron full down to the full up "fail safe" position was consistently measured to be eight seconds. The aileron null position change due to a channel failure was measured to be less than ½ degree.

5.5.2 Flight Test Results - DCI System

5.5.2.1 Test Flight No. 1

The mission of this flight was to perform the FCF, inflight EMI tests and the structural resonance test up to the maximum velocity for the aircraft.

The duration of the flight was one hour. The FCF was successfully completed. The EMI test and the structural resonance testing was conducted with a technical observer in the chase aircraft monitoring the left aileron. The observer did not detect uncommanded transients or sustained oscillations. A review of the recorded data confirmed the lack of either problem. The pilot commented that the test system flew much like the normal control system.

5.5.2.2 Test Flight No. 2

The mission of this test flight was to conduct system performance testing. Because of the limited time for the flight test left before having to give up the aircraft, an abbreviated set of test points (reducing from 33 to 15 the test points of Table 2) was planned in order to ensure obtaining data over the entended range of airspeed, inputs and altitude. The duration of the flight was .5 hours. The mission proceeded as planned, conducting lateral control performance tests through test point 14 of the planned 15.

During test point 15, the master caution light illuminated along with the utility hydraulic system light, indicating a loss of the utility hydraulic system. The aircraft then made an emergency landing on the lake bed.

5.5.2.3 Test Flight 2 Utility Hydraulic System Failure Analysis

The aircraft was towed to the modification hanger for inspection to determine the cause of the utility hydraulic system failure. The hydraulic systems were energized at a low pressure and a large leak was noted at the direct drive servovalve forward end cap. The DDCV actuator was removed for a more detailed investigation.

Visual inspection of the test aileron control actuator disclosed the source of the hydraulic leak. The O-ring sealing the end cap and suspension housing had been extruded out of the interface area where the parts meet. The end cap was removed for a closer examination and measurement. The magnet cap was found to have a .02" corner-to-corner warpage. The servovalve end caps are ported to a single system return pressure. For the test system they were connected to utility return pressure.

The DCI DDCV actuator end caps were designed and proof tested to withstand 1,500 psi return pressure (per McDonnell Douglas F-4 aileron actuator specification). The bending of the end caps implied that pressure in excess of 1,500 psi had been present in the aircraft utility hydraulic system return line.

Since redesigning the magnet caps and/or structurally reinforcing the existing caps to withstand higher return line pressures was impractical within the few days remaining for flight testing before the aircraft had to be turned over to another program, the hydraulic leak established the end of the flight test program.

Further laboratory testing on the flight test actuator was conducted. The actuator was returned to WPAFB. It was determined that the rear end cap of the DDCV had experienced permanent deformation of .014 inches, although no leak occurred.

Both end caps were remachined flat and a laboratory test setup made which provided for varying the return line presure to the test actuator while maintaining 3,000 psi inlet pressure. The return line pressure was increased in 100 psi increments. The results of the test on the same end cap that had failed is the following:

Return Pressure	Effect on End Cap
1,500	- none
1,600	- none
1,700	- interface gap of .002 in. without permanent deformation
1,800	- interface gap of .004 in. without permanent deformation

1,900	-	interface gap of .005 in. with permanent deformation of .002 in.
2,000	-	interface gap of .010 in. with .005 in permanent deformation
2,100	-	O-ring started to extrude
2,200	-	O-ring extruded out of end gap with accompanying fluid loss

The results of this test confirmed that the test actuator had been exposed to pressure in excess of the required proof pressure of 1,500 psi and most likely was subjected to pressure of 2,000 psi.

As part of the investigation as to the cause of the utility leak, the left wing utility system return line pressure vs return was measured on the test aircraft up to the point where a differential pressure of 1,000 psi was obtained. The recorded data was as follows:

Flow in GPM	Pressure in psi
3.0	125
4.0	180
5.0	305
6.0	430
7.0	525
8.0	620
9.0	725
10.0	1,000

In addition, the flow due to the combined motions of the aileron, inboard spoiler and outboard spoiler (which operate simultaneously when the aileron actuator is retracting near 0 degree aileron deflection) was calculated. The calculated maximim flow is 13.48 GPM and is based on measured maximum surface rates and the corresponding actuator rates (3.63 inches/second retract rate for the aileron actuator, 8.09 inches/second extend rate for the outboard spoiler and 11.26 inches/second extend rate for the inboard spoiler). The actuator areas used for the calculation are 7.8 square inches for the aileron (retract drive area connected to return), 1.4 square inches for the outboard spoiler and 1.03 square inches for the inboard spoiler.

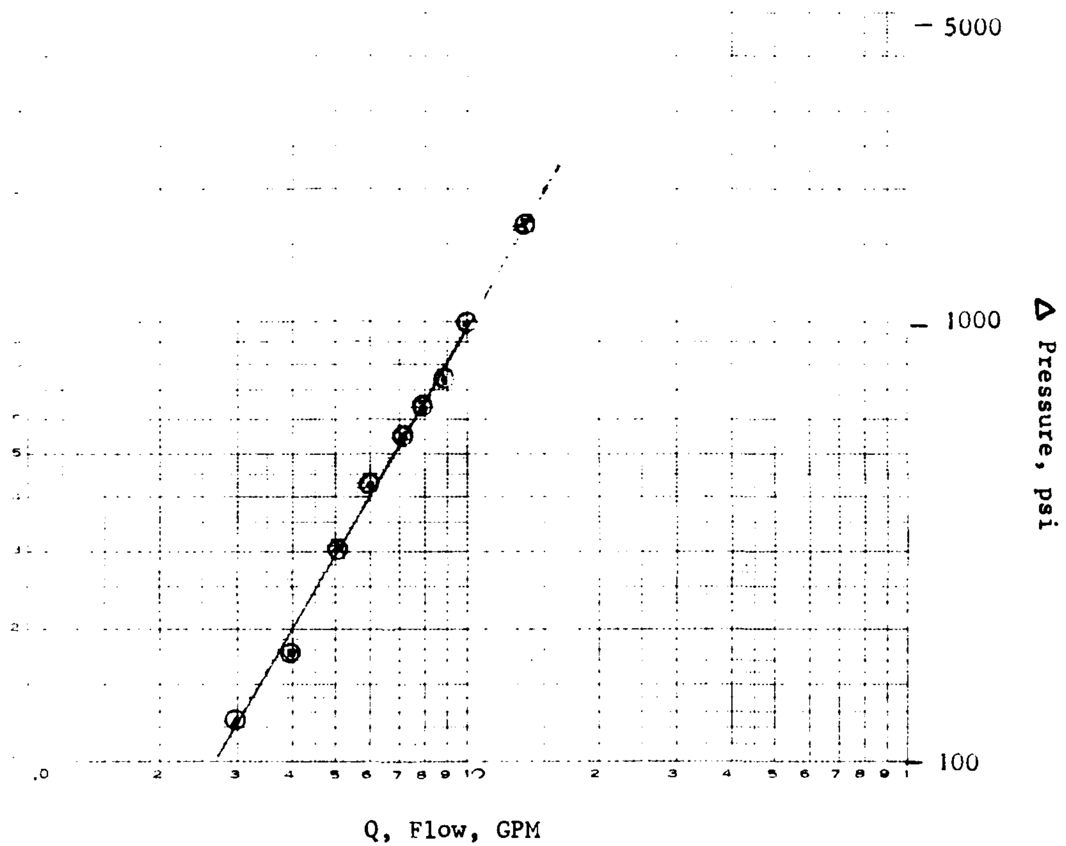
Figure 39 shows the measured data for the utility system return line and the calculated peak flow plotted on an extension of the line through the measured data. This figure indicates that the utility system would encounter a pressure drop of 1,650 psi steady state at maximum flow. Since several of the data acquisition test points required impulse type inputs or "stick raps", the system would have likely encountered transient pressure levels above the steady state value.

5.5.3 DCI Flight Test Data Analysis

The performance data obtained during the flight test of the DCI system provided only 15 of the 33 originally prescribed test points. However, the range of the test conditions provides a good comparison of the DDCV test system and the normal aircraft lateral control system performance.

Table 4 provides a summary of the DCI DDCV flight test data. The direction of roll is noted as either right, RT or left, LT. A right roll requires a right lateral stick input which in turn, directs the left aileron down to a maximum of 32 degrees, the right aileron up a maximum of 2 degrees, and the right spoilers up to a maximum of 42 degrees. The opposite sequence occurs when a left roll maneuver is commanded.

On Table 4, the airspeed and altitude is expressed as calibrated Mach number and feet above sea level. The maximum stick input amplitude, degrees right or left and stick input rate, degrees per second are listed. The stick input rate was measured at the main portion of the slope (ignoring the rounded courners at the start and stop positions).



⊙ Actual Data Point

⊗ Projected Data Point

Figure 39 P-Q Data for Utility Line From Left Aileron to Reservoir - A/C F4E #287

TABLE 4 DCI TEST DATA SUMMARY

Test point	Roll direction	Mach no.	Altitude (ft)	Max stick input (deg)	Stick input rate (deg/sec)	Maximum aileron position (deg)	Aileron extend rate (deg/sec)	Rate of roll (deg/sec/sec)	Maximum roll rate (deg/sec)
1	RT	.63	10,000	11.0	92.0	31	89 LT	240	180
2	LT	.63	10,000	11.0	82.0	28	87 RT	240	160
3*	RT	.63	10,000	11.0	73.0	32	80 LT	220	180
4	RT	.82	25,000	10.0	78.0	28	89 LT	230	200
5	LT	.82	25,000	9.0	73.0	23	100 RT	200	160
6*	RT	.82	25,000	10.0	70.0	10	84 LT	240	200
7	RT	.63	25,000	5.0	9.0	13	21 LT	40	70
8	LT	.63	25,000	7.0	10.0	14	15 RT	40	80
9	RT	1.27	25,000	10.0	68.0	31	70 LT	160	120
10	LT	1.27	25,000	10.0	59.0	22	64 RT	180	110
11	RT	1.30	40,000	2.5	1.5	6	3 LT	20	40
12	LT	1.30	40,000	3.0	6.0	10	10 RT	40	40
13	RT	1.30	40,000	9.5	61.0	25	80 LT	140	140
14	LT	1.30	40,000	9.0	48.0	21	78 RT	140	120
15	RT	.85	40,000	3.0	6.0	13	6 LT	40	40

* Channel 2 only of DDCV test system

In Table 4, the maximum aileron position attained is expressed in degrees of right or left aileron. The aileron extend rate expressed in degrees per second was measured at the major portion of the slope where the aileron traveled from neutral to a commanded down position.

As a measure of aircraft response, the rate of change of the aircraft roll rate in degrees per second per record is listed in Table 1 for each test point.

The gross weight of the aircraft was 42,589 lb at test point 1 and 36,889 lb at test point 15. The tests were conducted with the aircraft in a "clean" condition.

Test points 1, 2, and 3 provide a very good comparison of the aircraft response to a roll right command (with both one and two DDCV channels engaged) to a normal system's left roll response. Figures 40, 41, and 42 are excerpts from the recorded time history plots for test points 1, 2, and 3, respectively.

The maximum roll rate achieved for test point 1 (Figure 40) is 20 degrees per second greater than that recorded for test point 2, due to a surface deflection of 3 degrees greater for test point 1 than for test point 2 Figure 41). The velocities for the right and left aileron surfaces are comparable, even though there is a 10 degree per second difference between the stick input rates. The reason for this is that the surface rate is saturated with command inputs of greater than 70 degrees per second.

Figure 42 represents test point 3 with the DDCV system operating on the electrical channel 2 only. There is a slightly slower rate of change of aileron position with the reduced control system gain when operating on a single channel. The data beyond 2.5 seconds of time represents the aircraft being commanded back to straight and level flight. These three figures are representative of the test data obtained at the other test points.

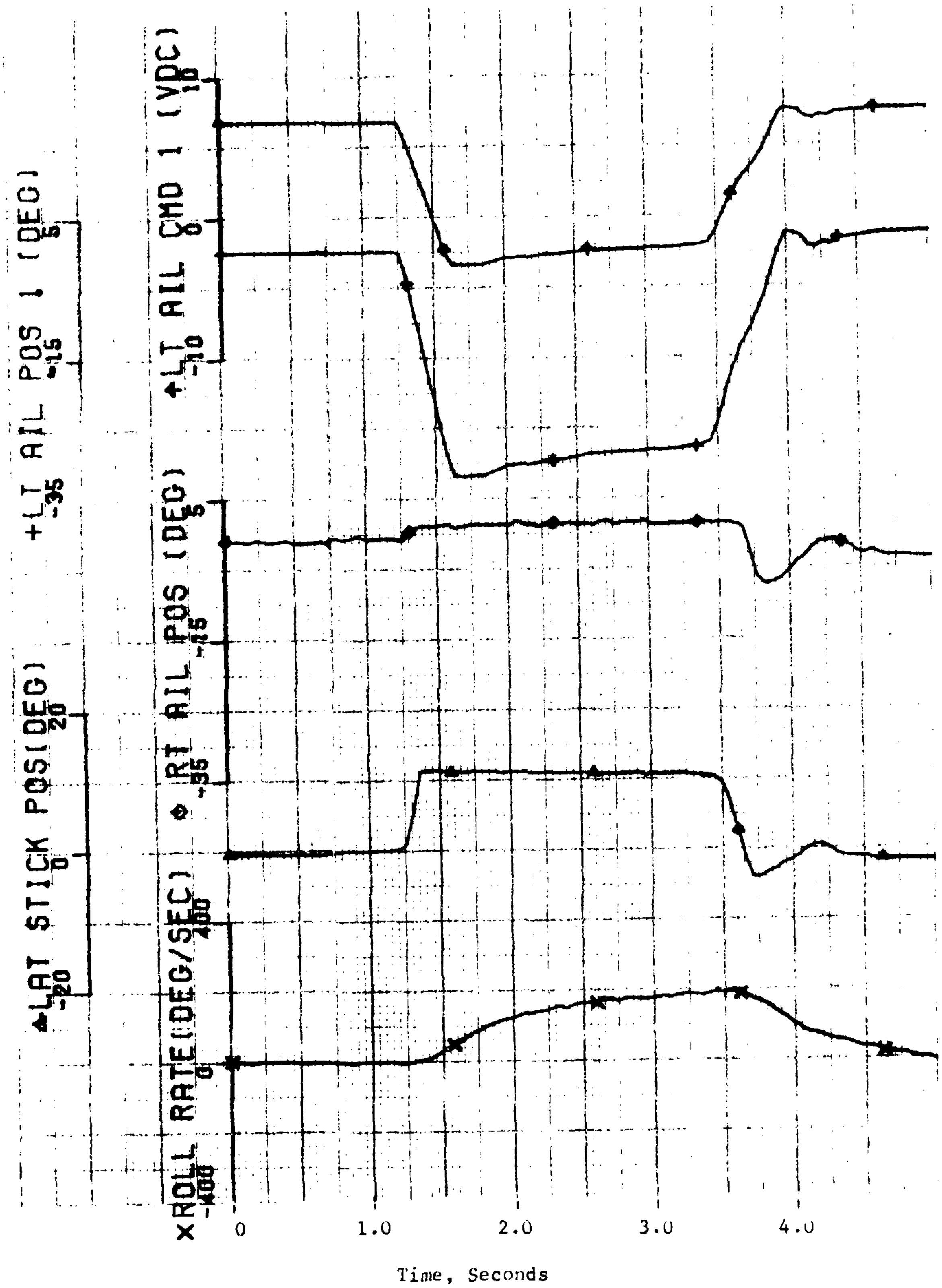


Figure 40 Flight Test Point #1

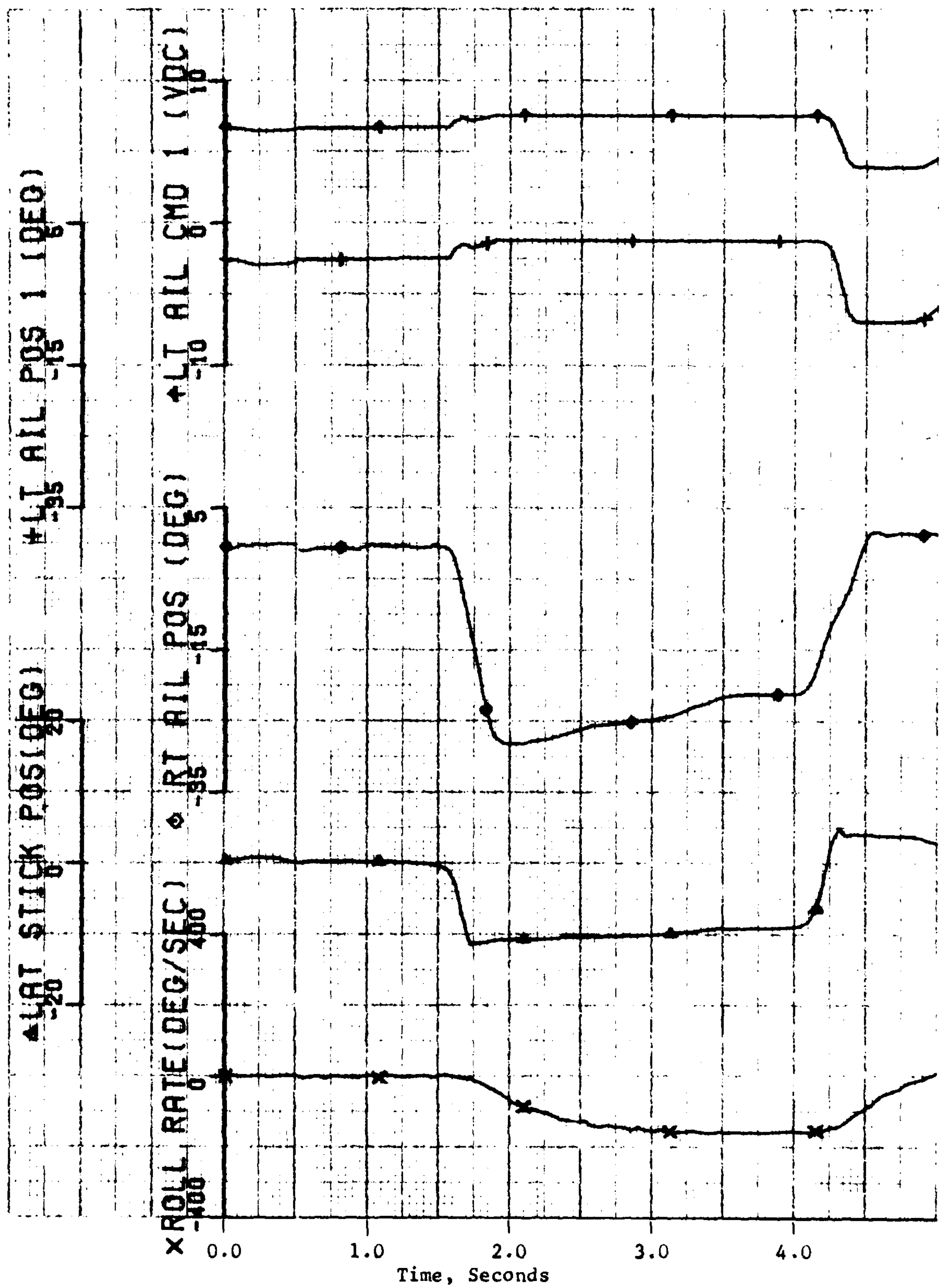


Figure 41 Flight Test Point #2

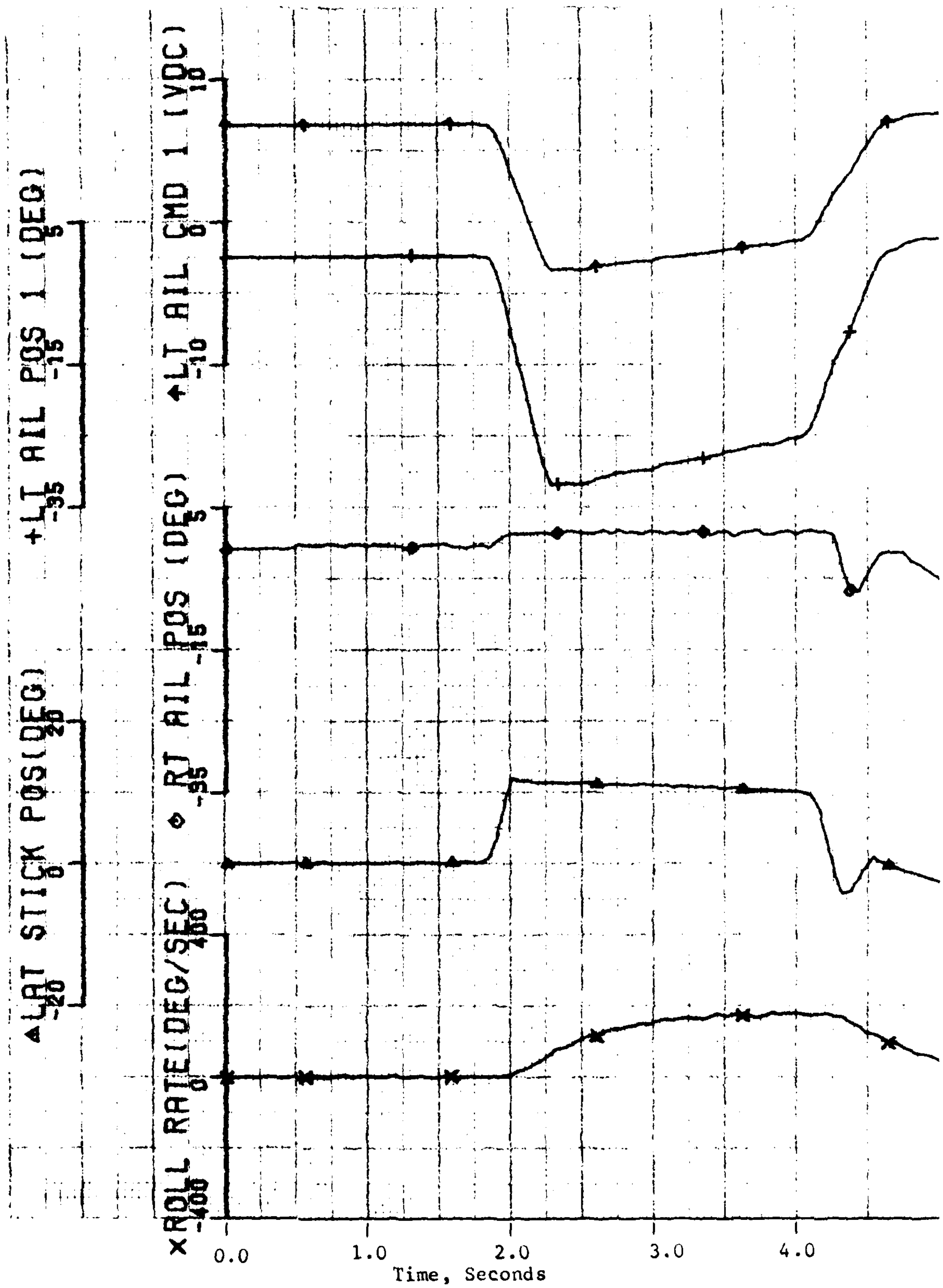


Figure 42 Flight Test Point #3

The data recorded verified that the test DCI DDCV system installed on the left aileron performed as well as the normal aircraft lateral control system. This statement applies to both single and two channel operation of the DCI DDCV system.

SECTION VI

RESULTS SUMMARY AND CONCLUSIONS

6.1 General

Both DDCV systems flew successfully. The operational flight time on the two systems was much shorter than originally planned because the time required for installation and ground check-out consumed most of the time available for the flight test program. The pilot's comments on both systems and the data recorded on the DCI system indicated that the operating characteristics of the two DDCV systems matched the aircraft's normal lateral control system well. Nusiance disconnects of the DDCV systems did not occur during the test flights (although a power interrupt did turn off one system when an interrupt time period beyond the 40 msec limit of the failure logic was encountered). Neither system was sensitive to EMI, either on the ground or in the air.

6.2 Lessons Learned

There were several general technical aspects to the program which should be noted for the purpose of avoiding problems in the future. These are the following:

1. If lag compensation is used in a FBW system in order to increase the low frequency gain of a control loop, reset and/or engagement transients can occur unless the design prevents the storage of DC voltages on the capacitors used to create the compensation network.

2. No "pre-installation" operating time was required on the flight hardware delivered for use in the test aircraft. In order to minimize using ground check-out time to "burn in" the flight hardware, an operational check-out of the actual flight hardware over the range of inputs and/or operating conditions (not just the nominal values) should be run in the laboratory. The operating time specified may be arbitrary, but on a flight test program such as the DDCV,

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FLIGHT TEST OF A G. E. AND DCI DIRECT DRIVE FLY-BY-WIRE FLIGHT --ETC(U)
JUN 82 G D JENNEY, H W SCHREADLEY F33615-78-C-3609
UNCLASSIFIED AFWAL-TR-82-3035 NL

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the "burn in" operation could have been easily as long as the planned flight test time without incurring significant cost. Problems discovered with the test hardware in the laboratory are much easier to isolate and correct than the same problem when it occurs in the test aircraft.

3. The high return pressure encountered with the test aircraft utility system created a problem within the DDCV system. This problem could have been avoided by designing the development hardware to withstand system pressure in the return lines. The weight penalty for such an approach is small. For future programs involving hydraulic hardware in the F-4, the fact that the pressure levels in the return line of the utility (and perhaps PC 1 and PC 2) system in the wing are above 2,000 psi under some operating conditions should be noted and considered in the hardware design.

6.3 Conclusions and Recommendations

Based upon the flight test results, it is concluded that there are no performance characteristics which inherently limit the application of direct drive control valve Fly-By-Wire systems for current aircraft. Both the G. E. and the DCI direct drive control valve actuators fit into the space for the normal manual input aileron actuator for the F-4, pointing out the packaging size advantage that the direct drive approach has over the secondary actuator approach to Fly-By-Wire control systems. (The small size of the direct drive control valve package can be a significant advantage in meeting the envelope restrictions for the control actuators of new aircraft.)

Both the G. E. and DCI direct drive systems performed satisfactorily, once the problems encountered during ground check-out were solved. The differences in the way the two systems were mechanized (moving iron vs moving coil force motors, LVDT's vs potentiometers, spool position feedback vs no spool position feedback) did not impact on the test data obtained. Either system appears to be a viable direct drive control mechanization.

It is recommended that direct drive control valve systems be considered on any new aircraft as a viable method of mechanizing a Fly-By-Wire Flight control system. It is also recommended that continued development work on direct drive control valves systems be conducted in order to incorporate technology improvements in magnetics, sensors and control electronics as they become available.

APPENDIX I

EMI TESTING

1. EMI GROUND TESTS WITH GROUND POWER

For this test a ground auxiliary power unit, a ground hydraulic Power unit, and an air-conditioning unit for equipment cooling will be connected to the aircraft. The aircraft data system will be turned on and an instrumentation ground support unit, preferably with strip chart capability, will be used to monitor the test system for electrical fluctuations. The left aileron, while stationary, will be monitored for sudden changes in position while the following EMI checks are made:

1.1 Turn on instrumentation ground power switch.

1.2 Turn Inertial Navigation System switches to STBY. Wait for HEAT light. Turn off. Repeat.

1.3 Generator control switch to EXT ON. Turn off. Repeat. Turn to EXT ON again.

1.4 Pull right transformer rectifier circuits breakers. Circuit breaker panel No. 2, 6A, 6B, 6C. Repeat for left rectifier 5A, 5B, 5C. Reset.

1.5 Pull circuit breakers for right fuel boost pump, 1E, 1F, 1G and reset. Pull circuit breakers for left fuel boost pump, 2E, 2F, 2G and reset.

1.6 Pull 28 volt transformer rectifier circuit breakers. Panel 2, 3C and 7A one at a time and reset.

1.7 Pull equipment cooling circuit breakers: Panel 3, 6B. Reset.

1.8 Cycle left engine master switch, On-Off. Cycle right engine master switch On-Off. Cycle both engine master switches On-Off. (Aircraft must be supplied with source of AC power for this test.)

1.9 Cycle anti-skid switch, On-Off.

1.10 Cycle engine anti-icing switch, De-Ice-Normal.

1.11 Cycle the flaps/slats switch, Out-Out and Down-Normal.

1.12 Cycle speed breaks, Out-In.

1.13 Cycle the CADC switch, Correction Off-Correction Reset-Normal.

1.14 Actuate rudder trim switch, Left-Right.

1.15 Cycle air refuel switch, Extend-Retract.

1.16 Cycle refuel select switch, Internal Only-All Tanks.

1.17 Cycle yaw, roll, and pitch stab aug switches, Engage-Off.

1.18 Cycle external lights switch, Taxi Light-Landing Light-Off.

1.19 Cycle flight instrument lights control, Bright-Off.

1.20 Cycle left generator control switch, Ext On-Off. Cycle right generator control switch, Ext On-Off. Cycle left and right generator control switches, Ext On-Off.

1.21 Transmit on the UHF radio, frequencies: 225.00, 260.00, 295.00 330.00, 365.00 and 399.95.

1.22 Interrogate the IFF in each of its modes with the ground interrogation unit.

1.23 Operate the TACAN.

1.24 Operate the radar altimeter.

1.25 Cycle the arresting hook handle, Down-Up.

1.26 Cycle pitot heat switch, On-Off. (Limit pitot heat operation to a maximum of one minute.)

1.27 Temperature control panel, switch to manual and cycle temperature selection knob from cold to hot.

1.28 White flood light switch cycle, On-Off.

1.29 Cockpit lights cycle, Bright-Off.

1.30 Cycle formation light, Join Up-Bright-Medium-Dim-Off.

1.31 Cycle exterior lights, Bright-Off, in both flash and steady modes.

1.32 Cycle the switch on the emergency floodlight panel, Bright-Off.

1.33 Activate nose wheel steering.

1.34 Activate pitch and roll trim.

1.35 Activate seat adjustment switch.

1.36 Place the test tool (referred to as the "spoon") in the right main gear scissor switch so the systems that operate when weight is off the right main gear will operate. The aircraft fuel pressurization valves and the angle of attack probe heaters should energize.

If EMI exists, then:

1.37 Pull the angle-of-attack circuit breakers 6C and 7C on the No. 3 circuit breaker panel, and also pull the Central Air Data Computer circuit breakers 1K, 2K, 3L, 4L, and 5L on the No. 4 circuit breaker panel to deactivate the angle-of-attack probe heaters. Reset.

1.38 Remove the test tool from the right main gear scissor switch.

1.39 Slowly cycle the left aileron and perform steps 11, 12, 14, 15, 17, 21, 22, 23, 24, and 34 and monitor the aircraft systems for fluctuations caused by electromagnetic interference of the test aileron system.

2. PART II EMI GROUND TESTS WITH AIRCRAFT POWER

Record time and hit the event button on the trigger switch the required number of times prior to each test point. For example, "E2" means hit the event button twice.

2.1 Perform a normal engine start. Set both engines at idle. E1

2.2 Turn Inertial Navigation System switches to STBY. Wait for HEAT light. Turn off. Repeat. E2

2.3 Generator control switch to EXT ON. Turn Off. Repeat. Turn to EXT ON again. E3

2.4 Pull right transformer rectifier circuits breakers. Circuit breaker panel No. 2, 6A, 6B, and 6C. Repeat for left rectifier 5A, 5B, and 5C. Reset. E4

2.5 Pull circuit breakers for right fuel boost pump, 1E, 1F, 1G and reset. Pull circuit breakers for left fuel boost pump, 2E, 2F, and 2G and reset. E1

2.6 Pull 28 volt transformer rectifier circuit breakers: Panel 2, 3C, and 7A one at a time and reset. E2

2.7 Pull equipment cooling circuit breakers: Panel 3, 6B. Reset. E3

2.8 Actuate rain removal switch. E4

2.9 Cycle anti-skid switch, On-Off. E1

2.10 Cycle engine anti-icing switch, De-Ice-Normal. E2

2.11 Cycle the flaps/slats switch, Out-Out and Down-Normal. E3

2.12 Cycle speed brakes, Out-In. E4

2.13 Cycle the CADC switch, Correction Off-Correction Reset-Normal. E1

2.14 Actuate rudder trim switch, Left-Right. E2

2.15 Cycle air refuel switch, Extend-Retract. E3

2.16 Cycle refuel select switch, Internal Only-All Tanks. E4

2.17 Cycle yaw, roll, and pitch stab aug switches, Engage-Off. E1

2.18 Cycle external lights switch, Taxi Light-Landing Light-Off. E2

2.19 Cycle flight instrument lights control, Bright-Off. E3

2.20 Cycle left generator control switch, Ext On-Off. Cycle right generator control switch, Ext On-Off. Cycle left and right generator control switches, Ext On-Off. E4

2.21 Transmit on the UHF radio, frequencies: 225.00, 260.00, 295.00, 330.00, 365.00, and 399.95. E1

2.22 Interrogate the IFF in each of its modes with the ground interrogation unit. E2

2.23 Operate the TACAN. E3

2.24 Operate the radar altimeter. E4

2.25 Cycle the arresting hook handle, Down-Up. E2

2.26 Cycle pitot heat switch, On-Off. (Limit pitot heat operation to a maximum of one minute.) E2

2.27 Temperature control panel, switch to manual and cycle temperature selection knob from cold to hot. E3

2.28 White flood light switch cycle, On-Off. E4

2.29 Cockpit lights cycle, Bright-Off. E1

2.30 Cycle formation light, Join Up-Bright-Off, Medium-Dim-Off. E2

2.31 Cycle exterior lights, Bright-Off, in both flash and steady modes. E3

2.32 Cycle the switch on the emergency floodlight panel, Bright-Off. E4

2.33 Activate nose wheel steering. E1

2.34 Activate pitch and roll trim. E2

2.35 Activate seat adjustment switch. E3

2.36 Place the test tool (referred to as the "spoon") in the right main gear scissor switch so the systems that operate when weight is off the right main gear will operate. The aircraft fuel pressurization valves and the angle of attack probe heaters should energize. E4

If EMI exists, then:

2.37 Pull the angle-of-attack circuit breakers 6C and 7C on the No. 3 circuit breaker panel, and also pull the Central Air Data Computer circuit breakers 1K, 2K, 3L, 4L, and 5L on the No. 4 circuit breaker panel to deactivate the angle-of-attack probe heaters. Reset. E1

2.38 Remove the test tool from the right main gear scissor switch. E2

2.39 Slowly cycle the left aileron and perform steps 11, 12, 14, 15, 17, 21, 22, 23, 24, and 34 and monitor the aircraft systems for fluctuations caused by electromagnetic interference of the test aileron system.

Use the following event code and record time prior to each step of this test point.

Event	Step
E3	11
E4	12
E1	14
E2	15
E3	17
E4	21
E1	22
E2	23
E3	24
E4	34

APPENDIX II

LATERAL RESPONSE EVALUATION PROCEEDURES, WITH TEST AILERON FAILED

1. During the initial Functional Check Flight of the DDCV aircraft the following procedures will be used to conduct a lateral control response evaluation of the aircraft.

1.1 System Reset Function Test. The DDCV system will be tested to insure proper reset function. Channel One of the system will be disabled by pulling the two appropriate circuit breakers. The circuit breaker will then be reset and the channel will be brought back on line with the Channel One reset switch. If this test is successful then Channel Two of the system will be tested in a similar manner. If both channels function properly in the reset test then both channels will be disabled by pulling the circuit breakers and a lateral control response will be conducted.

1.2 Lateral Control Response Evaluation. All tests will be performed at 15,000 feet MSL.

1.2.1 At each set of conditions the control response test will involve:

a. Establish and maintain straight and level flight.

b. Establish 15 deg. of right bank (30 deg. for points 1 and 2) then roll back to level flight.

c. If 1.2.1.2 above shows adequate lateral control then 15 deg. of left bank (30 deg. for points 1 and 2) will be slowly established and the aircraft will be rolled back to level flight.

1.2.2 The conditions for the lateral control response tests are given in Table 5. The test will not proceed to a subsequent test point if control at the previous point was inadequate. Control response will be considered adequate if, in the pilot's judgement, the aircraft can be safely flown while maneuvering to and performing a straight-in approach to a full-stop landing.

TABLE 5 TEST CONDITIONS

TEST POINT	SPEED/AOA (KIAS/UNITS)	CONFIGURATION
1	300/-	Cruise (Gear up, slats in, flaps up)
2	230/-	Cruise (Gear up, slats in, flaps up)
3	-/17	P.A. (Gear down, slats in, flaps up)
4	-/19.2	P.A. (Gear down, slats in, flaps up)
5	-/17	P.A. (Gear down, slats out, flaps up)
6	-/19.2	P.A. (Gear down, slats out, flaps up)
7	-/17	P.A. (Gear down, slats in, flaps down)
8	-/19.2	P.A. (Gear down, slats in, flaps down)
9	-/17	P.A. (Gear down, slats out, flaps down)
10	-/19.2	P.A. (Gear down, slats out, flaps down)

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